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Project Columbiad

Mission to the Moon

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VOLUME III
Stage Configuration Designs

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1 Mission Stages Overview

1.1 Mission Staging Profile

The EOR configuration for the piloted mission is composed of three propulsive elements in addition to the Crew Module: Primary Trans-Lunar Injection (PTLI), Lunar Braking Module (LBM), and Earth Return Module. The precursor mission is also composed of three propulsive elements in addition to its surface payloads: PTLI, LBM and the Payload Landing Module (PLM). Please refer to Volume I, Section 5.1 and 5.2 for a break-up of the different stages into the four launches. A quick summary: PTLI is on Launch 1 and 3 while the LBM, PLM, and surface payloads are on Launch 2 and another LBM, ERM, and CM on Launch 4.

The NLS vehicle does not perform the circularization burn into a 200 km altitude for any of the four launches. For Launches 1 and 3 the PTLI performs the circularization burn and then raises its altitude to 275 km at the desired trajectory window where it will await rendezvous with the piloted launch.

For Launches 2 and 4, the LBM performs both the circularization burn and the burn to higher orbit. Once the vehicles have completed rendezvous, the Trans-Lunar Injection burn is performed by two stages: the PTLI and the LBM. The PTLI separates from the stack upon the completion of its burn. The LBM completes the burn and then performs any midcourse corrections that are required during the 3 day transit. At which point the LBM inserts the vehicle into LLO, and then performs the major descent burn before it is staged.

For the precursor mission the PLM performs the final descent and hover burn before landing and deploying the habitat. A brief profile of the precursor mission along with propulsive requirements for each stage is featured in Table 1-1.

However, for the piloted mission, the ERM performs the final descent and hover burn before landing. After the 28 day lunar stay the ERM launches the CM into LLO and then into the Earth transfer orbit. The ERM also performs any midcourse corrections on the return trip. The ERM separates from the Crew Module (CM) just before reentry into the Earth's atmosphere and then the CM enters into the atmosphere. The piloted mission is completed when the CM lands at Edwards Air Force Base. A brief profile of the piloted mission along with propulsive requirements for each stage is also featured in Table 1-1.

Table 1-1: Mission Profile

Event	Location	Propulsive Stage(s)	ΔV (m/s)
Circularization of Launches 1 & 3	200 km LEO	PTLI	177
Launches 1 & 3 burn to higher LEO	200-275 km LEO	PTLI	43
Circularization of Launches 2 & 4	200 km LEO	LBM	177
Launches 2 & 4 burn to higher LEO	200-275 km LEO	LBM	43
Earth Orbit Rendezvous	275 km LEO	LBM & PTLI	60
Trans-Lunar Injection	LEO	PTLI	2460
Trans-Lunar Injection	LEO	LBM	680
Midcourse Corrections	Midcourse	LBM	120
Lunar Braking into LLO	Prior to LLO	LBM	1060
Lunar Braking to Moon	LLO to Moon	LBM	1700
Precursor Hover and Land	Moon	PLM	500
Piloted Hover	Moon	ERM	500
Lunar Launch	Moon to LLO	ERM	2200
Earth Return Injection	LLO	ERM	1060
Midcourse Corrections	Midcourse	ERM	120
Reentry	Earth's Atmosphere	CM	100

1.2 Commonality of Precursor with Piloted Vehicle

The precursor mission is designed to be as modular as possible with the piloted mission for developmental cost considerations. The first two stages of each, the PTLI and the LBM, are exactly the same to drive down the cost. As shown in Table 1-2, the velocity and masses are identical for each stage. Volume I, Sections 5.1.3 and 5.1.4, details the PTLI and LBM budgets.

Table 1-2: Commonality between Precursor and Piloted

	Precursor		Piloted	
Propulsion Stage	ΔV (m/s)	Mass (kg)	ΔV (m/s)	Mass (kg)
PTLI	2680	94,825	2680	94,825
LBM	3780	62,285	3780	62,285

2 Launch Vehicle Description

This chapter details the choice of launch vehicle for Project Columbiad, including descriptions of each of the vehicle's components, the configurations to be used for the precursor and piloted missions, launch facilities and schedules, and ascent trajectories.

2.1 Introduction - The National Launch System

Project Columbiad will utilize the National Launch System (NLS) as its launch vehicle. The NLS, considered by many to be the next logical step in the continuing development of a reliable American launch vehicle fleet, consists mostly of components derived from the Space Transportation System (STS). The NLS maximizes the use of existing technology, thereby minimizing development time and cost.

At the core of the NLS is a new engine derived from the Space Shuttle Main Engine (SSME), known as the Space Transportation Main Engine (STME). Four STME's are attached to the bottom of an extended External Tank (ET), also derived from the STS ET, forming the core of the NLS vehicle. Attached to the core are anywhere from two to four Solid Rocket Boosters (SRBs). These SRBs can either be the Redesigned Solid Rocket Motors (RSRMs) currently used for the Space Shuttle or, if the program is further funded, Advanced Solid Rocket Motors (ASRMs). The ASRMs provide almost the same thrust profile as the RSRMs, but fire for an additional 10 seconds.

The base configuration of the NLS originally examined for Project Columbiad uses two ASRMs providing a capability of approximately 72 metric tons (mt) to low earth orbit (LEO). For Project Columbiad, a capacity of 91 mt is required. While the National Aeronautics and Space Administration (NASA) is considering the development of new liquid rocket boosters to increase the payload capacity of the NLS, Project Columbiad seeks to limit additional development costs by increasing the number of SRBs on the vehicle.

Project Columbiad's piloted and precursor missions will each require two NLS launches, with rendezvous and docking operations in LEO to integrate the vehicle for Trans-Lunar Injection (TLI).

2.2 Launch Vehicle Configuration for Project Columbiad

This section discusses the components of the National Launch System and how they are combined to form the complete launch vehicle.

2.2.1 Launch Vehicle Components

This subsection describes the various components of the NLS, including the STMEs, core structure, and SRBs.

2.2.1.1 Space Transportation Main Engines

The STME is currently under development by Rocketdyne, Pratt and Whitney, and Aerojet. Like the SSME, the STME uses Liquid Oxygen (LO_2) for oxidizer and Liquid Hydrogen (LH_2) for fuel, with an oxidizer to fuel ratio of 6:1. It provides 2650 kN of vacuum thrust with a specific impulse (I_{sp}) of 430.5 s for a burn time of 416.5 s.

As shown in Figure 2-1, the STME is approximately 3.7 m long by 2.1 m in diameter, with an expansion ratio of 45:1. It has a dry weight of 3600 kg, and can be gimballed up to 8.5 degrees in any direction from the nominal thrust direction. This ability provides thrust vector control (TVC) and the use of load relief.

Finally, the STME can be throttled in single percent increments from 75% to 100%. The engines will be throttled to 75% in the area where maximum dynamic pressure is experienced and again near main engine cutoff (MECO) in order to reduce axial loading on the NLS vehicle.

It is currently estimated that the STME will have reached 99% reliability with 50% confidence testing by the time prototype flights are scheduled in 1998. [Colgrove, 1991]

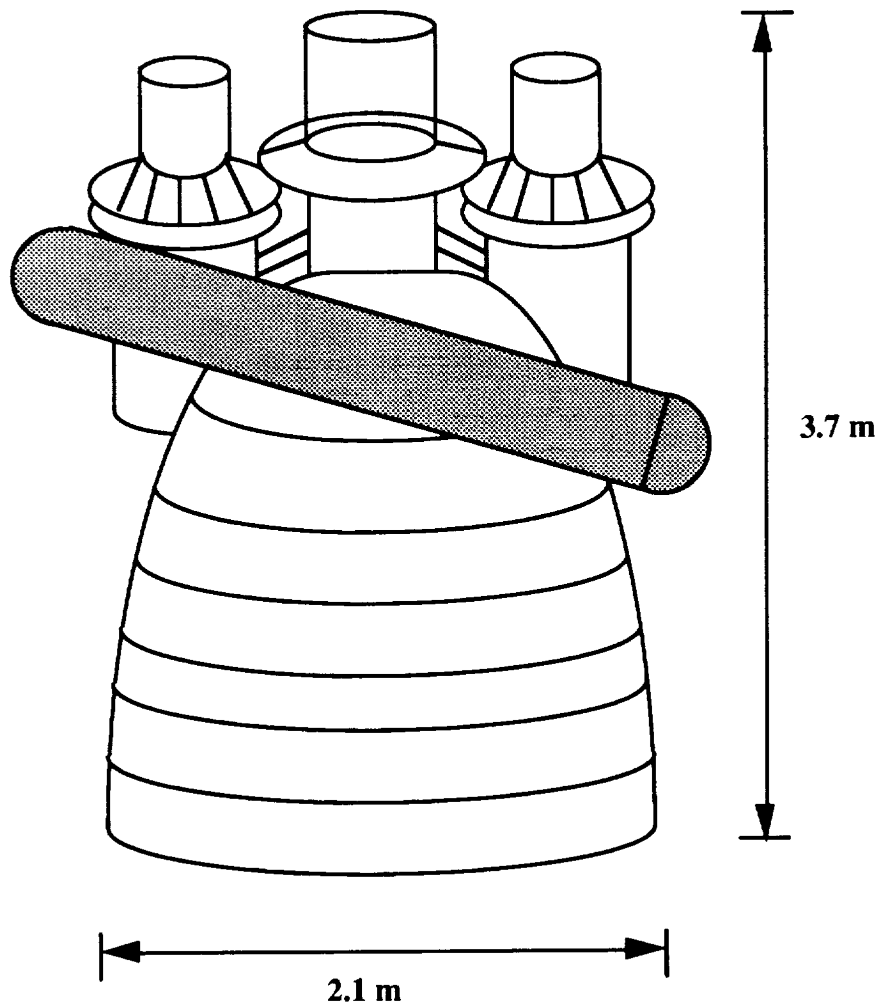


Figure 2-1
The Space Transportation Main Engine

2.2.1.2 Core Structure

The core structure of the NLS consists of a stretched version of a Space Shuttle ET. Its LH_2 tank is stretched approximately 1.5 m over the Space Shuttle version, so that the total length of the core is 48.5 m (plus engine boattail and payload interface section), with a diameter of 8.4 m.

The NLS Core Structure is shown in exploded view in Figure 2-2. Additional structural stiffness has been provided to the intertank section by the use of a crossbeam. In addition, the top cone of the Space Shuttle LO_2 tank has been replaced with a more familiar barrel top to accommodate the payload interface section above the tank. Finally, new feed lines are provided from both tanks down to the aft boattail, where they connect with the four

STMEs. As in the Space Shuttle, the LO_2 tank is equipped with baffles to reduce sloshing. Such baffles are not necessary in the LH_2 tank because of the fuel's low density.

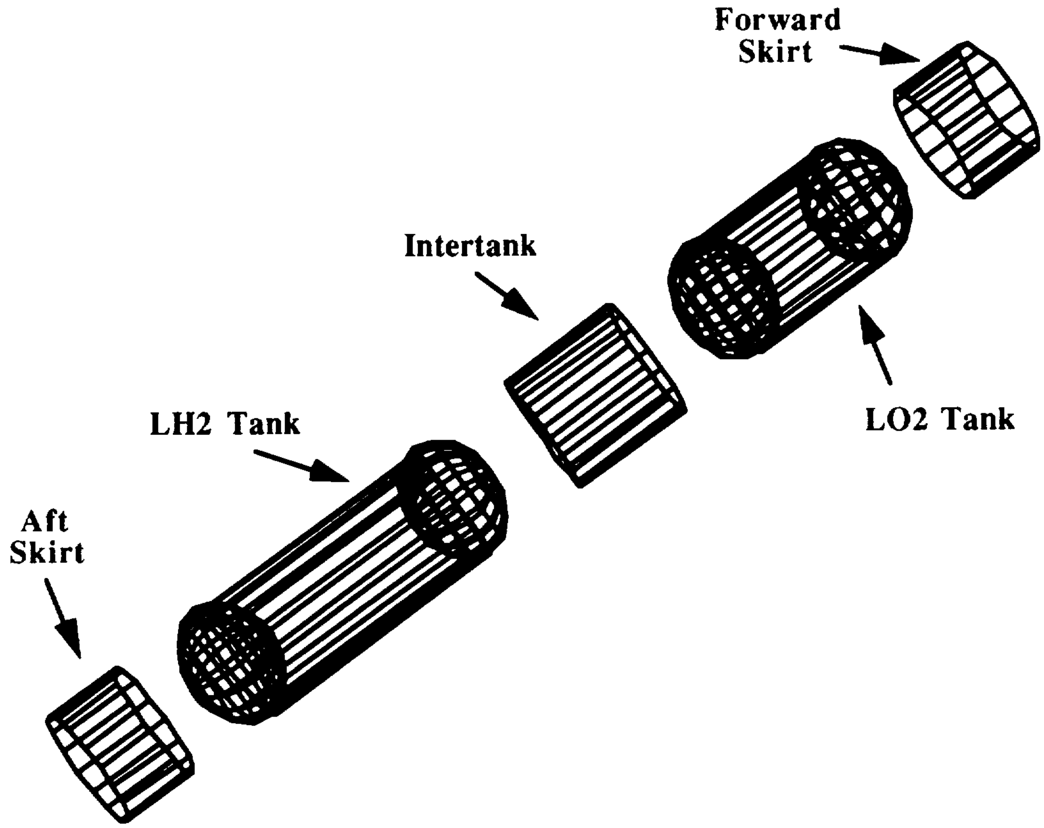


Figure 2-2
NLS Core Structure

Table 2-1 breaks down the mass of the core structure, including engine assembly. These numbers are adapted from a NASA NLS Reference System Definition document [NASA, 1991].

Table 2-1: NLS Core Structure Mass Breakdown

Item	Mass (kg)
Forward Structures	1538
LO_2 Tank	6389
Intertank	6036
LH_2 Tank	16592
Thermal Protection Sys.	1804
STME Assembly	14553
Feed System	4509
Pneumatic System	1453
Avionics	681
Attach & Separation	494
Subsystem Structure	1208
Thrust Structure	8349
Thrust Vector Control	1347
Contingency	6514
Total Core Dry Mass	**71467**

The fueled core vehicle carries a total of 766.4 mt of propellant; 109.5 mt of LH_2 and 656.9 mt of LO_2.

2.2.1.3 Solid Rocket Boosters

In all original considerations of the NLS, it was assumed that the SRBs used would be ASRMs. It now seems likely that funding for the ASRMs will be canceled, necessitating the use of Space Shuttle RSRMs. For this report, a description of both types of SRBs is provided. Should funding for the ASRMs be reinstated in the future, they may prove to be the better choice of SRB for Project Columbiad.

2.2.1.3.1 Redesigned Solid Rocket Motors

After the *Challenger* accident of 1986, the standard SRBs used for the Space Shuttle were redesigned to solve the now-famous "O-ring" problem. These motors, known formally as Redesigned Solid Rocket Motors (RSRMs), are now the standard for all Space Shuttle flights. Supplied by the Morton Thiokol Corporation, they provide a vacuum thrust level of 11760 kN with an I_{sp} of 270.3 seconds and a burn time of 123 seconds. The RSRM (without its nose cone) is 38.4 m long with a maximum diameter (at the nozzle) of 3.88 m.

The booster is assembled from four segments of solid fuel (insulated with asbestos), plus the nose cone and nozzle sections. The nose cone and the nozzle section each contain four separation motors which help to push the ASRMs away from the NLS vehicle after the solids have burned out. In addition, the nose cone contains a parachute so that the ASRM can be recovered by NASA trawlers after jettison.

The RSRM uses Polybutadiene-acrylic acid-actylonitrile terpolymer (PBAN) solid propellant, weighing approximately 503.3 mt. The inert weight of the motor case (made from D6AC steel) and nozzle totals 55.8 mt, and the separation motors and recovery systems weigh 11.9 mt. Therefore, the total launch weight of one RSRM is approximately 571 mt. [NASA, 1990] The assembled RSRM is shown in Figure 2-3 on the following page.

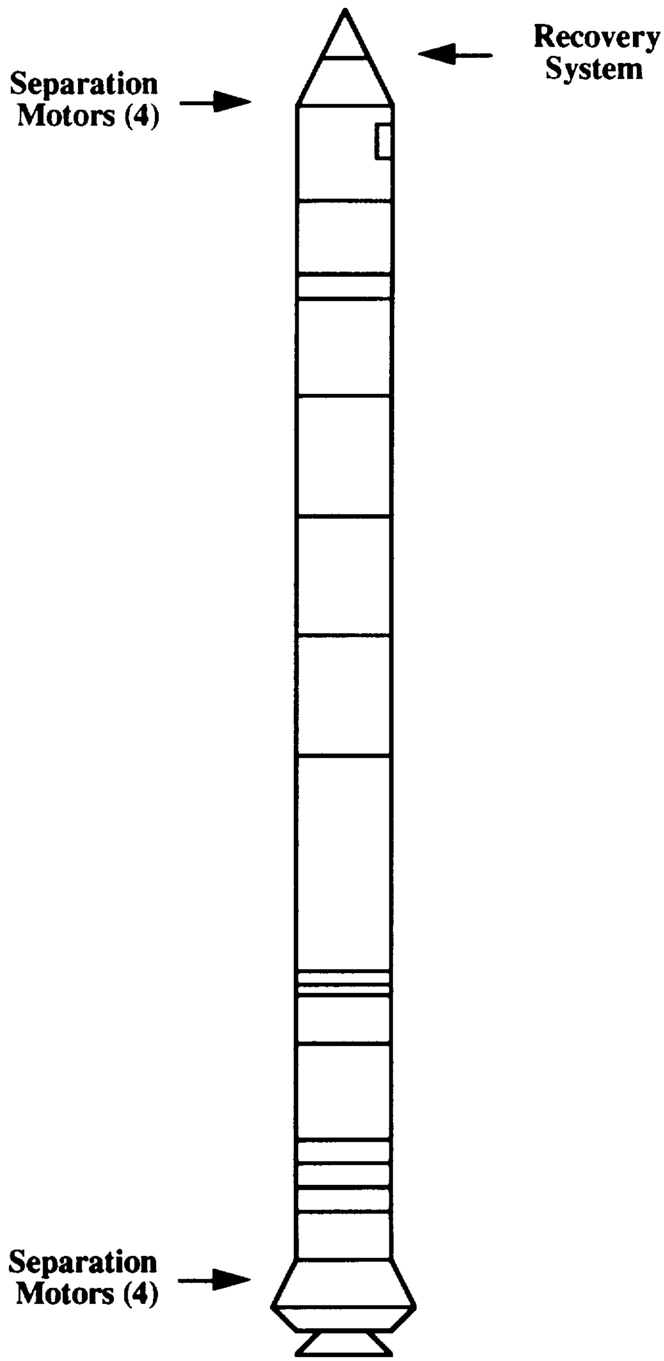


Figure 2-3
Redesigned Solid Rocket Motor (RSRM)

2.2.1.3.2 Advanced Solid Rocket Motors

The Advanced Solid Rocket Motor (ASRM) looks almost identical to the RSRM (see Figure 2-3), but has some interesting differences. The ASRM provides approximately

11900 kN of vacuum thrust with an I_{sp} of 270.3 s and a burn time of 134 s (compared to 123 s for the RSRM). It has the same length and approximately the same diameter as the RSRM (38.4 m and 3.81 m, respectively). This booster uses a three fuel segment design, with 548.1 mt of Hydroxy-terminated polybutadiene (HTPB) solid propellant. The motor is insulated with a combination of Kevlar and glass. The inert weight of the motor case (made from 9 Ni-4 Co-0.3 C alloy) and nozzle totals 52.9 mt, and the separation motors and recovery systems weigh 10.7 mt. The total launch weight of one ASRM adds up to approximately 611.7 mt. [NASA, 1990]

2.2.2 Launch Vehicle Integration and Configuration

The complete NLS vehicle consists of 4 STMEs, 2 to 4 SRBs, the payload interface, and the payload. The STMEs are arranged at the corners of a square on the bottom of the aft boattail, as shown in Figure 2-4. The SRBs are attached to the NLS vehicle both at the intertank section and at the top of the aft skirt. At SRB separation, the attach points are severed with small explosive charges before the separation motors are fired.

2.2.2.1 Launch Vehicle Footprints and Payload Configurations

Figure 2-4 shows possible vehicle footprints with 2, 3, and 4 SRBs. The 3 SRB configuration shown may be more difficult to achieve than the other two configurations for two reasons. First, it yields an asymmetric thrust profile. Second, it reduces clearance between the STMEs and the SRBs. It may be possible to reconfigure this model so that the SRBs are located at the vertices of an equilateral triangle.

For reasons of payload capability and thrust symmetry, it was decided to use the 4 RSRM configuration for Project Columbiad.

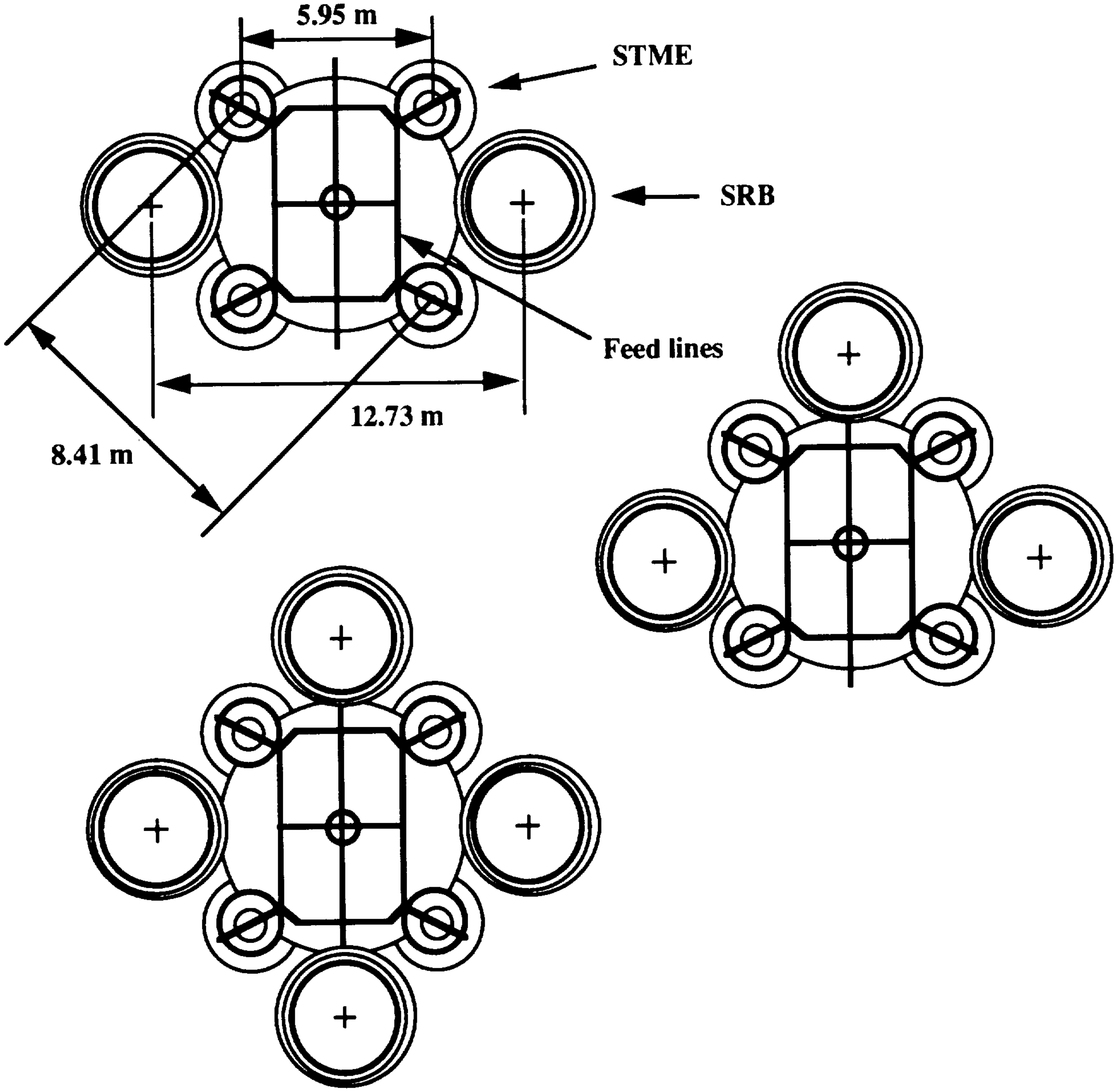


Figure 2-4
NLS Vehicle Footprints

The NLS vehicle will be assembled vertically in one of the high bays in the Vehicle Assembly Building (VAB) at Kennedy Space Center (KSC). The completed launch vehicle is shown in Figure 2-5.

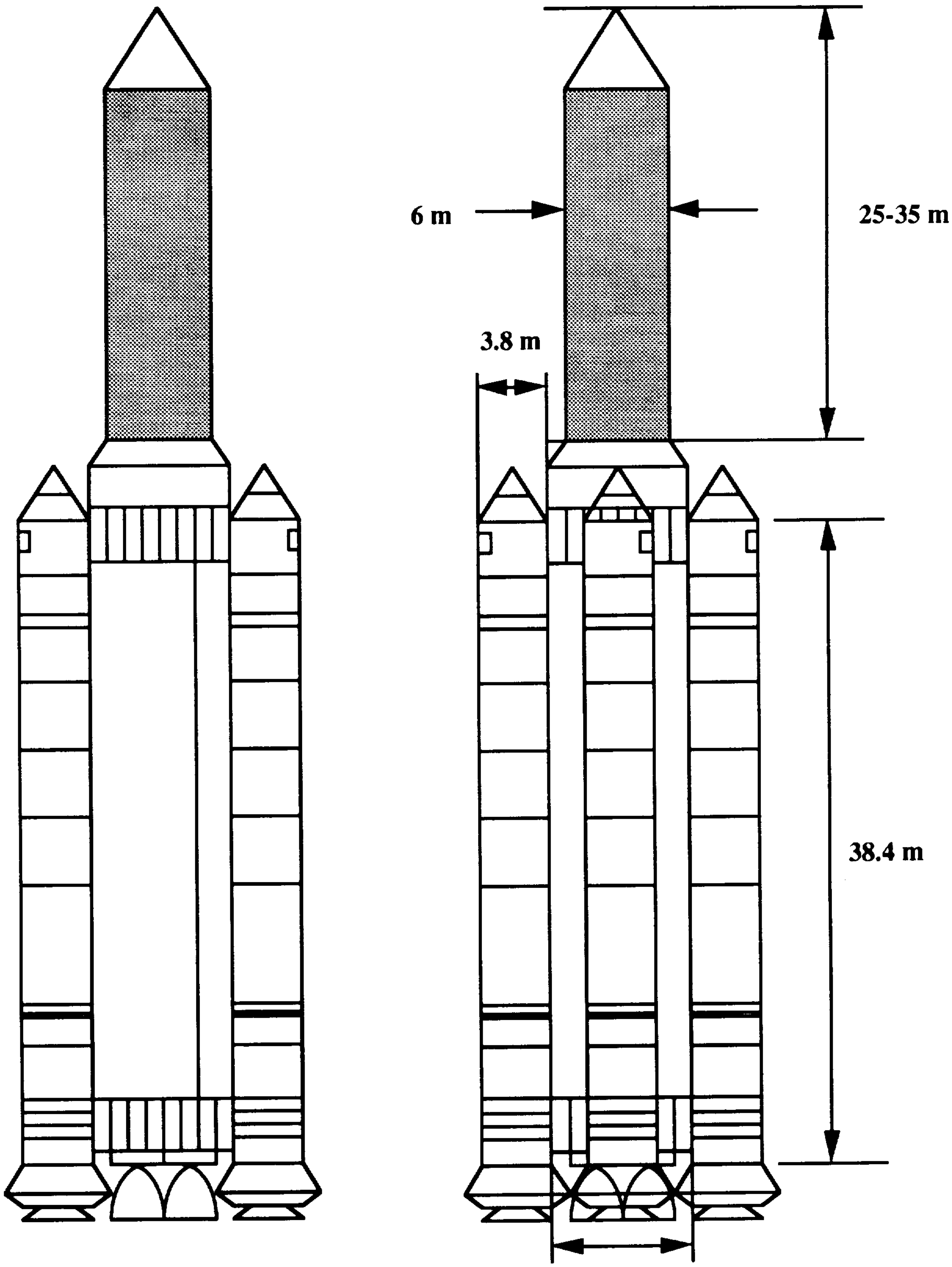


Figure 2-5
NLS Launch Configurations

The gray areas in Figure 2-5 represent the actual payload stacks. The stack for the piloted mission is shown in Figure 2-6, and the precursor stack is shown in Figure 2-7 .

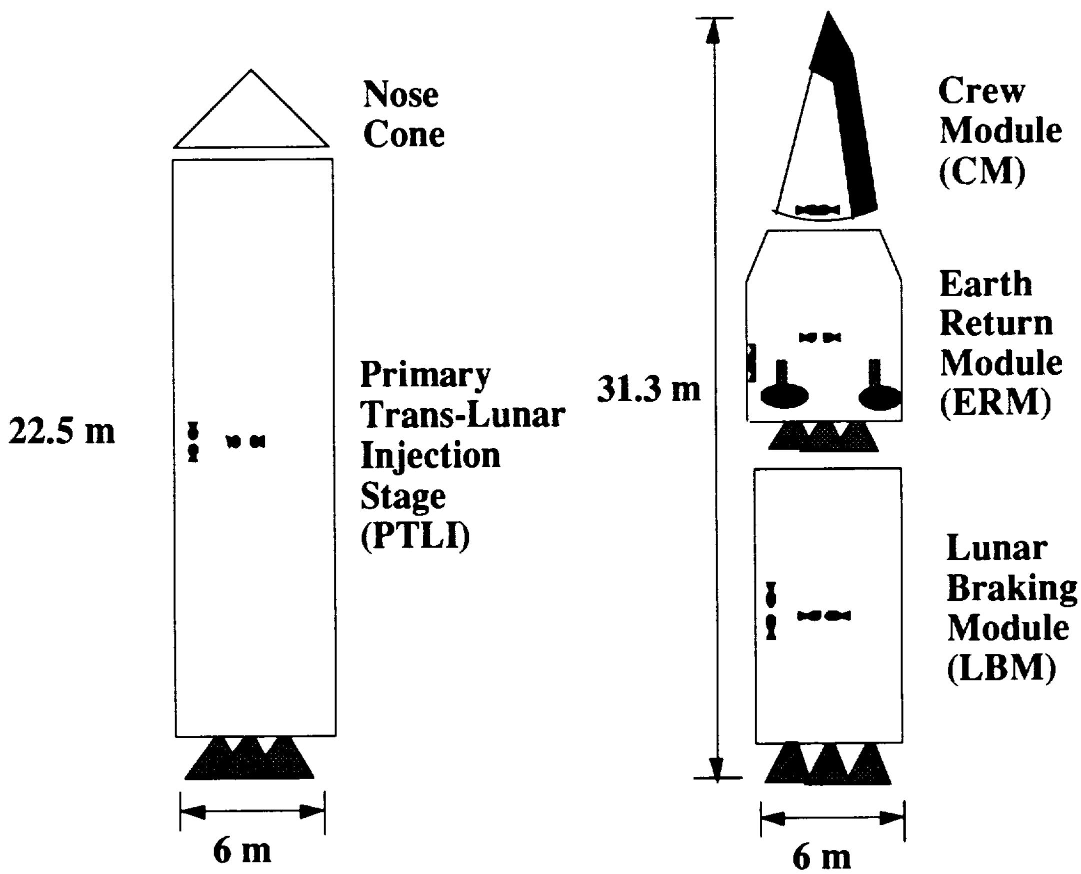


Figure 2-6
Payload Stacks for Piloted Mission

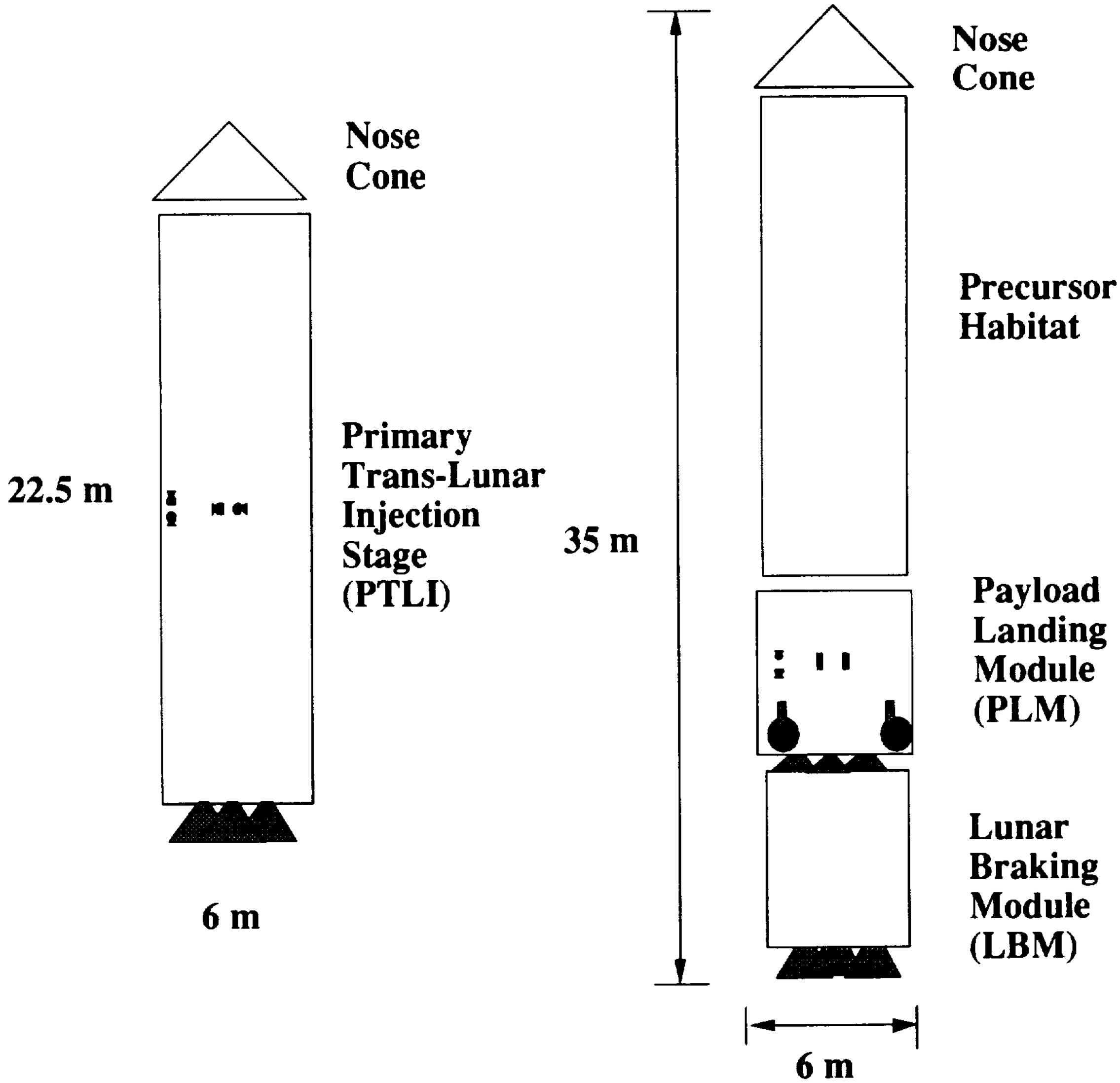


Figure 2-7
Payload Stacks for Precursor Mission

2.2.2.2 Fairings and Payload Interfaces

The fairing used on the launch vehicle depends on the payload. For both precursor launches and the PTLI launch of the piloted mission will use a nose cone to provide an aerodynamic profile for atmospheric flight. This nose cone is made of aluminum, 5 m tall with a maximum diameter of 6 m. It weighs approximately 820 kg, and contains a small explosive charge which separates the cone from the rest of the launch vehicle in the upper atmosphere. [NASA, 1991] The separation system for the nose cone is the same as for the payload interface, as explained below. (See Figures 2-8 and 2-9.) The piloted vehicle, because of its biconic shape and due to abort considerations, will have no nose cone.

The interface section will be a 4.5 m long aluminum and composite stiffened structure connecting the forward skirt of the core vehicle and the aft section of the payload. This section transitions from the 8.4 m core diameter to the 6 m payload diameter and houses the aft section of the PTLI (or LBM) engines and launch vehicle instrumentation ring (Figure 2-8). The aluminum skin will have composite stiffened panels to support the mass of the payload under all axial, lateral, and torsional g-conditions. The composite stringers will decrease the mass of the interface section which is now estimated at 1750 kg.

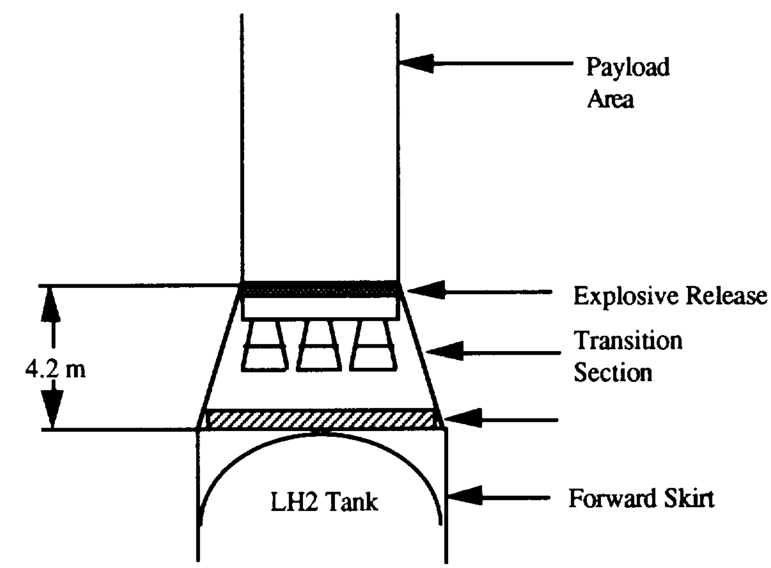


Figure 2-8
NLS Payload Interface

The upper section of the interface structure contains an explosive release ring within the wall of the vehicle (Figure 2-9). When the core vehicle is staged from the payload section, a small explosive charge is detonated which, in turn, causes expansion of the connection ring and separation of the core vehicle. This method allows for a smooth separation with minimum explosive force. Redundant systems may be employed as required.

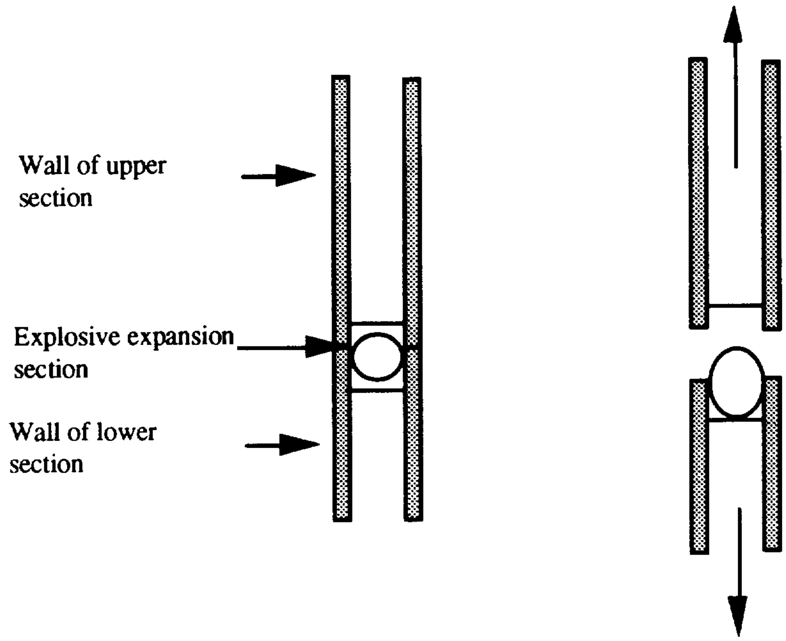


Figure 2-9
Explosive Separation System

The interface section also houses the launch vehicle guidance and instrumentation ring which are jettisoned with the core vehicle. This ring is estimated to be 0.75 m tall with a wall depth of 0.5 m. The total instrumentation weight and structure is estimated at 350 kg.

2.2.2.3 Payload Capabilities

Table 2-2 compares the capabilities of the three NLS configurations studied for Project Columbiad. Each configuration has an engine out capability throughout the ascent, with an additional payload capacity of about 3 mt if all engines function normally. In the event of an engine failure, that engine's fuel is redistributed and burned among the remaining three STMEs.

Table 2-2: NLS Configuration Comparisons

# of SRBs	Payload (mt)	Max axial g's
2 ASRMs	72	4.0
3 ASRMs	83	4.0
4 RSRMs	91	4.0

Notice that Table 2-2 assumes the use of a kick motor for orbit circularization after MECO. For Project Columbiad's 4 RSRM configuration, no kick motor will be used, so the payload capacity approaches 100 mt. The gross vehicle lift-off weight for this configuration is approximately 3227 mt, depending on the specific payload.

2.2.3 Projected Reliability

Table 2-3 provides estimates of NLS reliability based on historical data of the systems from which it is derived. The figures presented are for a 4 STME system, with a one engine-out capability, and are taken from a 1991 presentation to the National Research Council. [L Systems, 1991]

Table 2-3: Failure Probability of NLS Components

System	Failure Probability
RSRM	0.010
STME (Benign)	0.000
STME (Catastrophic)	0.004
Stage Level	0.002
Engine-Out Control	0.002
Guidance	0.002
Other Subsystems	0.005
Total	**0.025**

Because the individual failure probabilities are so small, it was decided that a reasonable estimate of the overall probability can be obtained by summing them. Therefore, the overall system reliability is estimated to be 97.5%. However, because data was limited, it was estimated in the L Systems document that this figure may fall anywhere between 96% to 98.5%.

Unfortunately, these numbers are for a "mature" system, i.e. more than 100 flights, well into the 21st century, given the currently planned launch frequency. In fact, it is quite possible that the system reliability would not break 90% before the tenth flight. Clearly, the system would not be fully matured within the time frame of Project Columbiad. This leads to two choices: delay the lunar mission until the NLS has been more thoroughly flight tested, or increase the number of ground-level tests until it hurts and hope to reduce the risk of failure that way.

2.3 Launch Facilities: The Kennedy Space Center Launch Complex

The NLS vehicle will be assembled and launched from the Kennedy Space Center (KSC). The vehicle will use modified Shuttle launch pads 39A and 39B as well as other complex facilities. Other NASA centers involved with the launch and control of the mission are the Johnson Space Center (JSC), the Goddard Space Flight Center (GSFC), the White Sands tracking and communications facility and other worldwide tracking stations.

2.3.1 Launch Complex 39: Possible Modifications

The NLS is a system based on existing Shuttle hardware and facilities. The NLS will require minimal KSC launch complex modifications. It is important that the operations of the NLS not hinder the Space Shuttle's capability. The capability for simultaneous Shuttle assembly, testing and launching is currently possible and must be extended to the NLS for the purposes of Project Columbiad.

Launch Complexes (LC) 39A and 39B were originally constructed for the Saturn V but are currently being used for Shuttle launches. Some modifications were made to the pads to support a vigorous Shuttle schedule.

The launch tower has been modified and a payload transfer structure constructed. This structure swings into place for payload transfer, then swings back out before launch. The transfer structure is used in place of the mobile access tower utilized for the Saturn V. This structure is highly specialized for the Shuttle and would not help the NLS because the payload is stacked, rather than side mounted, and will be integrated in the Vehicle Assembly Building (VAB). This structure may be modified to support the NLS as an access tower.

Figure 2-10 [Benson, 1978] shows LC 39A with the existing fuel storage and transfer facilities, crawlerway and flame deflector. No modification to the power, fuel or pressurization facilities at LC 39A will have to be done for NLS launches. Note that from the top view of LC 39A, one can distinguish the areas where the ejection seats may be used once the SRBs are ignited. The best orientation for the capsule is such that the ejection seats fire straight back along the crawlerway which is free of structures and obstacles.

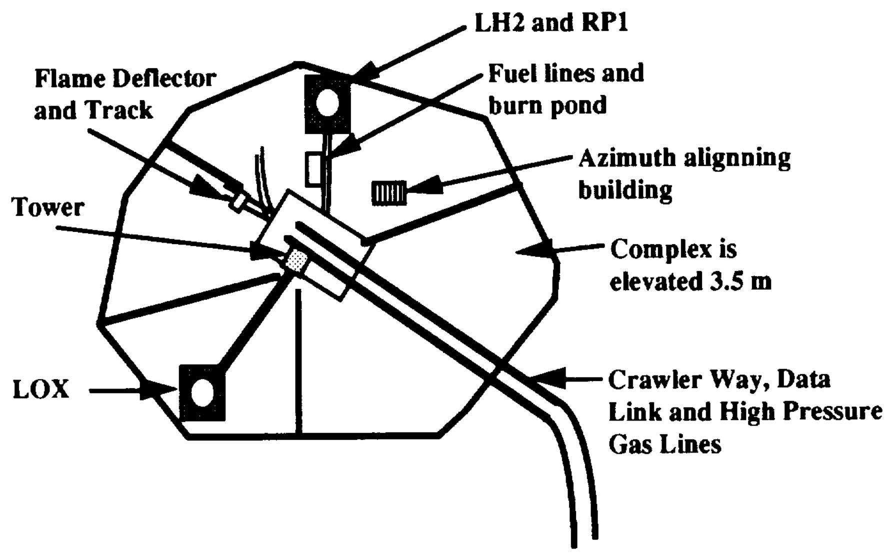


Figure 2-10
Launch Complex 39A

No structural modifications to the launch complex, mobile launch pads, or crawler would be necessary as the weights of the current NLS configurations are commensurate with that of the Saturn V. LC 39 modifications will include an extension of the fixed service structure, modifications to the exhaust channel of the mobile launch pad, and modifications to the Saturn V mobile access tower or extension of the payload transfer structure for use as an access tower.

The fixed service structure at LC 39 will have to be reconstructed or extended for the NLS. Figure 2-11 shows the necessary modifications to LC 39. The tower will be approximately 125 m tall with two stability arm locations, one at the current height for the Shuttle on the pad, and one at the height of the NLS core on the pad. These arms are used to keep the vehicle stable while fueling, and are released 60 seconds before launch. All modifications will be done in such a way as to allow for both Shuttle launches and launches of various NLS configurations.

The vehicles are assembled on the mobile launch pad in the VAB. The mobile launch pad is then moved to the launch complex via the crawlerway. These pads have built-in exhaust channels and vehicle hold down points. The mobile pad for the NLS vehicle configurations will have different hold down points and exhaust channels depending on the configuration. The NLS four-booster configuration (NLS-4B) will have to have a pad very similar to that used for the Saturn V vehicle. (See Figure 2-12.) The mobile launch pads used for the NLS will have one large exhaust channel for the extra SRBs as opposed to the Shuttle's three exhaust channels.

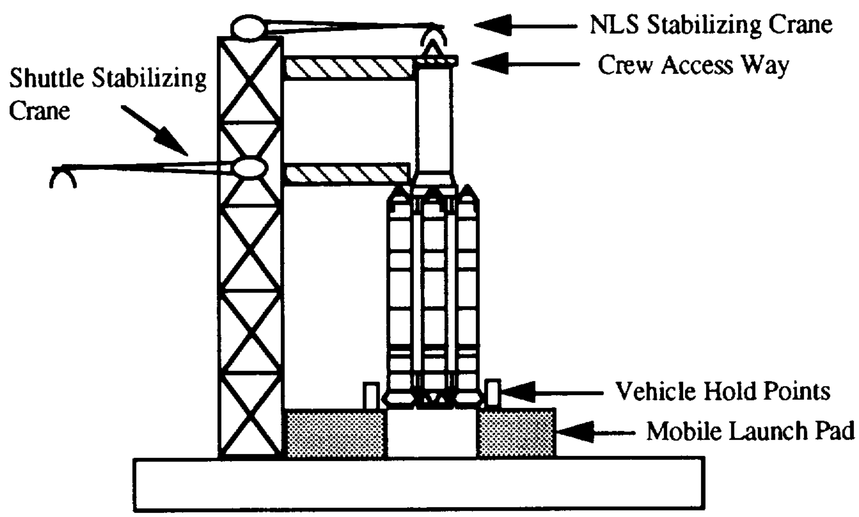


Figure 2-11
Modifications to the LC 39 Fixed Service Structure

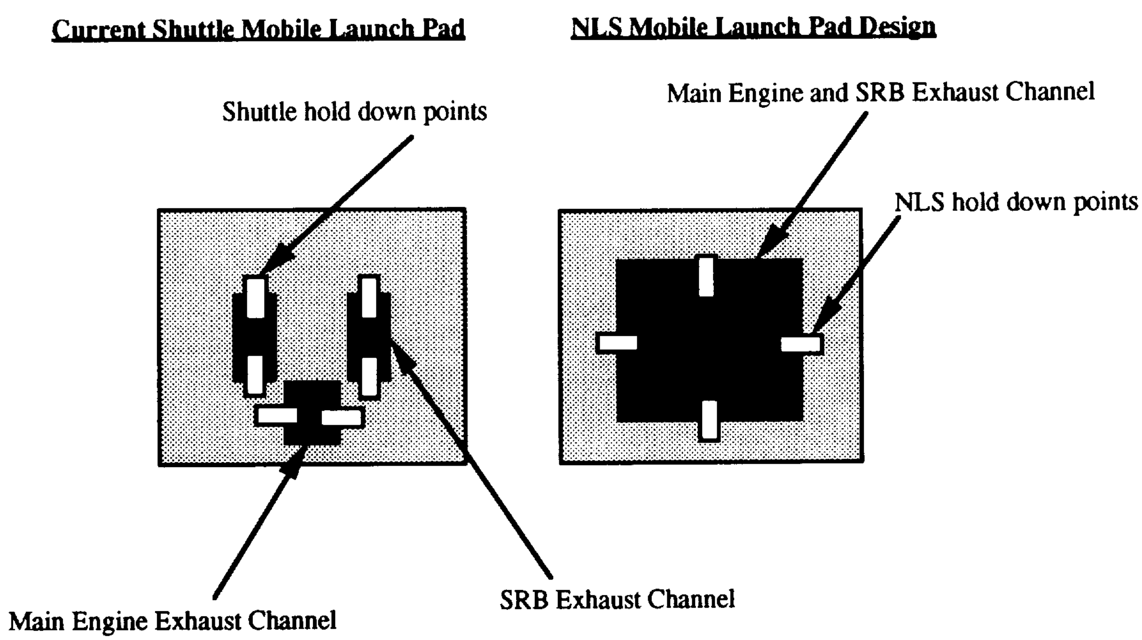


Figure 2-12
Comparison of Shuttle and NLS Mobile Launch Pads

It may be necessary to reconstruct the mobile access tower used for the Saturn V vehicle. This structure is pulled up to the pad after the vehicle is in place and is used as an access and service tower. The payload transfer structure may be modified and used in this way. The location for ordnance and vehicle test points will drive the access tower design.

The Vehicle Assembly Building (VAB) was constructed for the simultaneous checkout and assembly of four Saturn V vehicles. It measures 160 m high, 156 m deep, and 205 m wide [Bilstein, 1980]. The VAB is currently used for Space Shuttle assembly. It will be possible to assemble two NLS vehicles without hindering the Shuttle assembly. Although the VAB has expansion capability, no modifications will be necessary in the early stages of the program.

2.3.2 Ground Facilities, Support, and Safety

Adjacent to the VAB is the Launch Control Center (LCC). This control center houses all of the prelaunch, launch, and post launch automation, guidance, tracking, and telemetry computers and personnel. The LCC is in direct and constant communication with the Integrated Mission Control Center (IMCC) in Houston. The LCC retains vehicle and launch control until the vehicle has cleared the tower (t=6 sec) at which point control is turned over to the IMCC. The LCC continues to track, send guidance information, and telemeter status information as a backup check.

The NLS will utilize the Tracking and Data Relay Satellite System (TDRSS). This system uses two geostationary relay satellites 130° apart in longitude along with the White Sands complex and other ground facilities allowing full communication, tracking and telemetry. The NASA Communications Network (Nascom) managed by the Goddard Space Fight Center (GSFC) forms the ground links between tracking stations, the IMCC, and the LCC.

Launch personnel and launch complex safety is the task of the KSC Office of Safety, Reliability, Quality Assurance and Protective Services. This office oversees the shipping, receiving, testing, cleaning, assembly, fueling, transportation, and launch of the vehicle and its payload. Dangerous work situations or equipment are reported to the safety office and dealt with in a procedural manner. The safety office incorporates the Range Safety Office (RSO).

The RSO deals with range safety at each stage of prelaunch, launch, and abort. First during fueling and countdown the RSO dictates the necessary spacing distances for fuel lines, fuel storage, equipment, personnel and observers. LC 39 range safety was designed for a catastrophic explosion of the Saturn V vehicle such that none of the facilities, fuel lines or ground staff bunkers would be damaged. A blast overpressure study for the NLS was completed for the LC 39 integration as well as for abort success analysis for an explosion during ascent. The results are similar to those for the Saturn V but reveal a much diminished liquid fuel risk as a result of the solid motors.

The analysis was done using a TNT equivalent model for the liquid fuel in the NLS vehicle. The overpressure from TNT explosions is well documented and shows that the pressure drops radially as $1/r^3$. The following is a break down of the formulation [MIT, 1990].

NLS fuel mass (@ t=0) M = 766.4 mt

TNT conversion factor C = 0.2
(experimentally derived)

TNT potential energy E = 5413 KJ/kg

$$\textbf{Pressure} = (M \times C \times E) / r^3 \tag{2-1}$$

The theory breaks down near the center of the blast but conforms to experimental measurements up to about a 50 meter radius. The temperature at the core can reach 1600 °C and 500 ° C at the outer edges of the propagating blast front. A fully fueled Saturn V first stage was detonated in New Mexico for the purposes of defining the safety range needed at the launch site [Benson, 1978]. Detailed information about the test are not available, but the results show that they had overestimated the blast pressure.

The plot on the next page (Figure 2-13) shows the blast pressure for the fully fueled NLS-4B. This does not take into account the hazards from the SRB deflagration or from any other fuel source on board. The blast pressure will decrease linearly with decreasing fuel and atmospheric pressure and will reach zero at MECO.

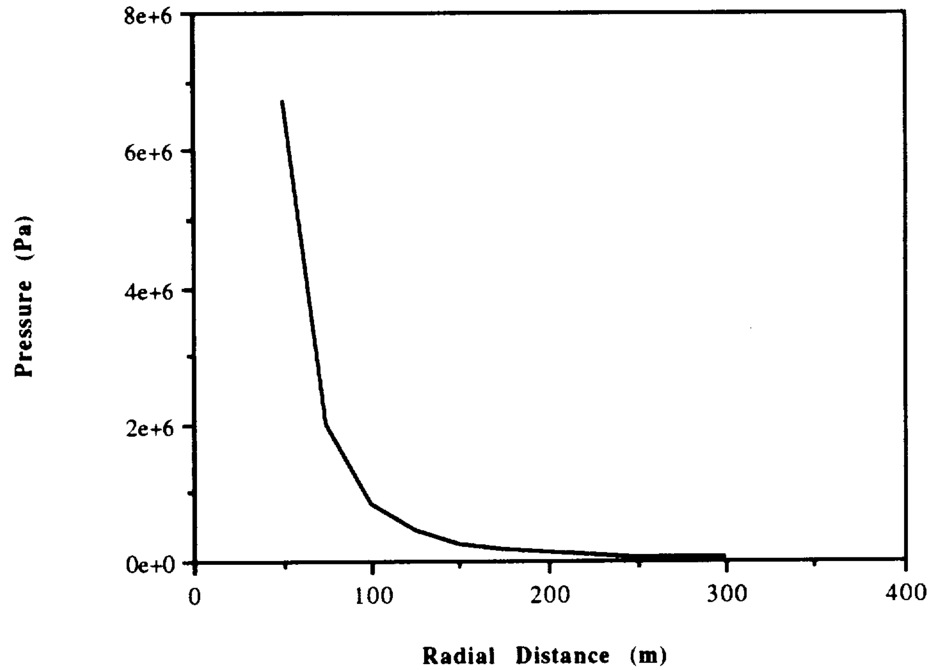


Figure 2-13
NLS Blast Overpressure

After ignition, the range safety officer surveys top and side camera views of the launch vehicle as it rises next to the tower. If the vehicle begins to drift into the tower the officer will notify LCC who will then take proper measures. Once the tower is cleared, range safety is responsible for jettisoned components and abort maneuvers.

During a normal flight the range safety officer will track the jettisoned SRBs and core structure and predict their flight paths. The SRB splashdown area is cleared before launch while the NLS core burns up over the Indian ocean. The RSO has set the requirement that all powered stages of a vehicle must have destructive ordinance with double redundancy. The range safety officer observes a real time trajectory plot superimposed on a destruct zone. If the projected point of impact strays outside of the destruct zone, the safety officer sends the detonation signal. Range problems may also arise in the case of an abort in which the SRBs or core structure are jettisoned on a trajectory that places them in the destruct zone. In this case, the range safety officer sends an arming signal to the receivers on the vehicle which in turn initiates thrust termination, then the destruct signal [Benson, 1978] .

2.4 Piloted Mission Launch Events

This section describes events relating specifically to the two NLS launches for Project Columbiad's piloted mission.

2.4.1 Launch Window and General Launch Schedule

One project goal of Columbiad is to enable a lunar landing at any location on the lunar surface at any time of the year. However, once a particular landing site is specified, there is only one available launch window each month (approximately). Depending on the landing site, this window length may range from less than a day to over four days. This must be included in the final determination of the launch schedule.

The piloted mission will require two launches to place the required amount of mass into LEO. The Primary Trans-Lunar Injection stage (PTLI) will be launched first into a 200 km circular orbit. From that altitude, a Hohmann transfer will be performed, leaving the stage in a 275 km circular orbit. The PTLI stage will remain in this orbit until rendezvous and mating with the piloted payload. The piloted payload will be launched second, into the same 200 km orbit, with the Hohmann transfer burn time chosen for closest approach to the PTLI stage at 275 km.

As with the precursor mission, the delay between these two launches is flexible. The minimum wait would be the time for one orbit of the PTLI stage, to insure it has achieved the required 275 km circular orbit. In the event that a stable orbit for the PTLI stage is not achieved within the mission window to the Moon, the piloted launch can be delayed until the next launch window opens. The PTLI stage would maintain its 275 km circular orbit for the duration of this delay. This schedule is shown in Table 2-4 below.

Table 2-4: Piloted Launch Order, Delays, and Considerations

Launch Order	Payload	Delay of Launch	Considerations
1	PTLI stage	(none)	Maintain orbit
2	Piloted	90 min - 1 month	Launch Window

2.4.2 Vehicle Assembly

The processing and assembling process for the piloted mission will closely follow that which is used by the Space Shuttle program, as will the precursor mission. The payload processing will be the same for the precursor payload and for both PTLI stacks. The Payload Processing Room (PPR) can be used for checkout and preparation of the crew capsule, and the entire vehicle will be assembled in the VAB.

The assembly of the launch vehicle occurs as its components arrive at the VAB. The SRBs will be stacked and aligned on the mobile launch platform, before they are attached to the NLS core vehicle. The piloted payload will then be stacked onto the vehicle and all interfaces connected and checked. The installation of ordnance devices (explosive bolts, separation and range safety charges) occurs at the pad. Using the Space Shuttle program as a guide, vehicle assembly should take place in 40 working hours. The move to the launch pad, making all connections, fueling, checkout, and launch should take a minimum of 24 working hours. The system should be capable of launch within two hours after the filling of the propellant tanks is started. [Kaplan, 1978]

2.4.3 Launch Sequence

Once the vehicle is secured on the launch pad, initial systems checks will be made. Upon completion, the final countdown will be ready to start and will follow the timetable shown in Table 2-5, adapted from the Space Shuttle program. [Joels, 1982] Launch Control is at KSC and Mission Control is at JSC. The sequence for launching the PTLI stage is exactly the same, omitting crew-specific events in the countdown.

Table 2-5: Piloted Launch Sequence

Time (Takeoff minus hr:min:sec)	Event
T - 5:00:00	Begin final countdown
T - 4:30:00	Begin filling liquid-oxygen tank in NLS Core & Payload Stages
T - 2:50:00	Begin filling liquid-hydrogen tank in Core & Payload Stages
T - 1:50:00	Enter crew capsule

T - 1:30:00	Communication-link checks with Launch Control
T - 1:25:00	Communication-link checks with Mission Control
T - 1:20:00	Abort advisory check
T - 1:10:00	Capsule hatch closure
T - 1:05:00	Cabin leak check
T - 0:51:00	Inertial Measurement Unit (IMU) preflight alignment
T - 0:50:00	Water-boiler and nitrogen supply preactivate
T - 0:32:00	Primary avionics software system/backup flight system (BFS) transfer prep
T - 0:30:00	Ground crew secures tower and retires to fall-back area Crew cabin vent
T - 0:25:00	Voice check Weather update
T - 0:21:00	Close vent valves
T - 0:20:00	Load flight plan into computers
T - 0:19:00	Load flight plan into BFS
T - 0:15:00	Abort Check
T - 0:09:00	1 minute hold to prep for final phase of countdown
T - 0:09:00	Resume countdown Go for launch
T - 0:07:00	All access arms retract
T - 0:06:00	Auxiliary power unit (APU) prestart
T - 0:05:00	Start APUs
T - 0:04:30	Capsule switches to internal power
T - 0:03:00	STMEs gimbal to launch positions
T - 0:02:55	NLS oxygen vents close Liquid-oxygen tanks begins pressurizing
T - 0:02:00	All systems configure for liftoff

T - 0:01:57	NLS hydrogen vents close Liquid-hydrogen tank pressures build
T - 0:00:25	SRB APUs start Countdown management switches to onboard computers
T - 0:00:03.8	Computers command STMEs to start
T - 0:00:03.46	STMEs begin to ignite in sequence
T - 0:00:00	Check STME pressure Check STME status 2.64 second timer for SRB ignition starts
T + 0:00:02.64	SRBs ignite
T + 0:00:03	Lift-off
T + 0:00:06	Launch tower cleared
T + 0:00:06	Roll and pitch maneuvers begin
T + 0:00:30	Roll maneuver completed

2.4.4 Launch Abort Modes

A general discussion at the launch abort modes was presented in Volume 1, as well as an in-depth discussion of the ejection seats which will be housed in the crew capsule and abort modes throughout the piloted mission. This subchapter will discuss the specific abort modes relating to the launch of the piloted mission.

There are six abort modes for the piloted mission launch. "Piloted mission launch" is defined as the time from the crew's entrance into the capsule until the post-MECO orbital insertion burn. The abort modes are as follows, and are discussed in subsequent sections: [Baker, 1985]

1. Redundant-Set-Launch Sequencer: from initial crew occupation up until SRB ignition.

2. SRB-Powered Flight Ejection: from SRB ignition up to when the vehicle reaches 36,500 m altitude (from launch until about T + 0:00:82).

3. Capsule Release and Ejection: for a time period beginning with the decline of the SRB thrust profile (approximately T + 0:01:17) to a predetermined trajectory point when there is enough energy and propellant to insure the success of abort mode number 4.

4. Trans-Atlantic Abort:: where there is insufficient propellant to insure the success of abort mode number 5. This can consist of either a capsule landing or an ejection.

5. Abort-Once-Around: where insufficient propellant remains to push on into an orbit.

6. Abort -To-Orbit: when sufficient propellant remains to send the capsule into orbit.

There is no scheduled abort mode for the 35 seconds between reaching 36,500m altitude and when the SRBs burn out. (See Volume I, Section 5.4.4.4) This is the time between abort mode numbers 2 and 3. There is not sufficient thrust in the Earth Return Module (ERM) to release the capsule from the NLS stack during this interval.

2.4.4.1 Redundant-Set-Launch Sequencer Abort

This type of abort will be used on the pad as long as the SRBs have not experienced ignition. It calls for cessation of the launch countdown, and for the crew to egress from the capsule via the access arm to the slide wire escape system mounted to the service structure. This is the same slide wire system currently used in the Space Shuttle system. It is illustrated in Figure 2-14 below. [Kaplan, 1978] There are five slide wires with one escape basket per wire. Each basket can hold up to two crew members, but the capsule's crew of four would only need to use four of the five wires with one person per basket. The basket slides into an arresting net after 35 seconds of travel. The astronauts can then enter a protective bunker.

Computer diagnosis of the launch vehicle is important during this abort mode. It occurs during a relatively brief time period where correct diagnosis and subsequent STME shutdown can save the mission before the SRBs ignite, thus forcing a launch in non-optimal conditions. This period of rapid, computerized, pre-SRB ignition diagnosis is extremely useful and vital to building mission success and operational reliability.

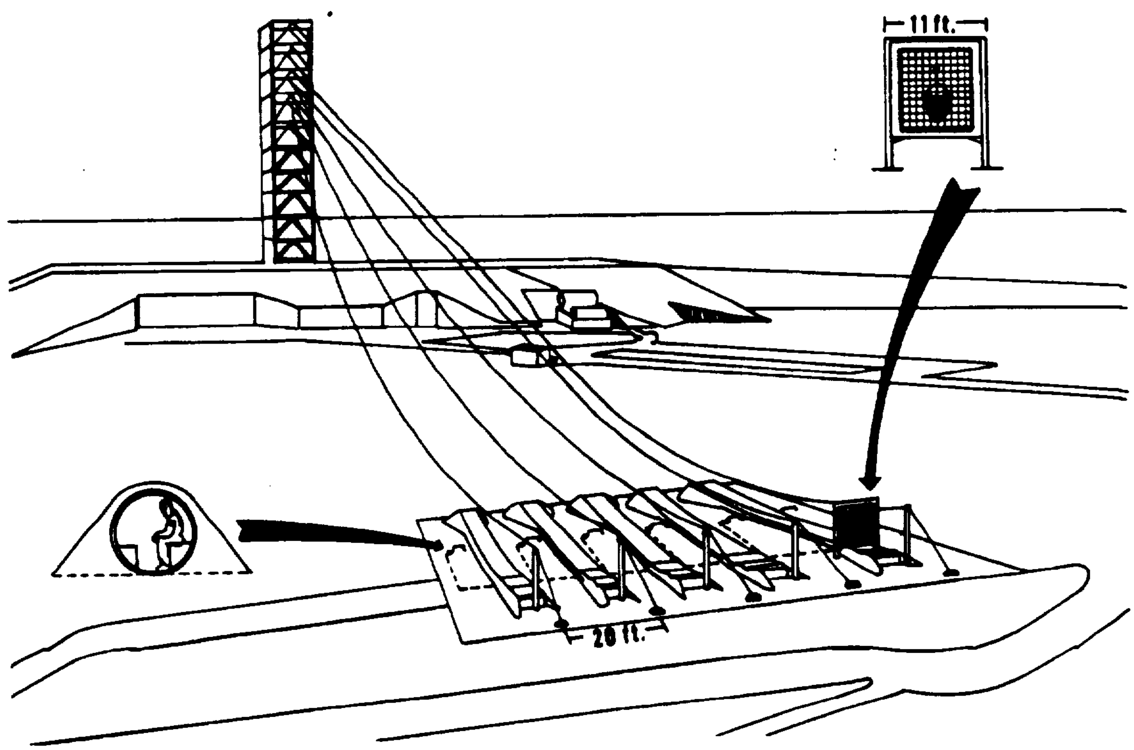


Figure 2-14
Slide Wire Escape for RSLS Abort

2.4.4.2 Booster Powered Flight Ejection Mode

This mode is to be used prior to the vehicle attaining 36,500 m altitude, while the SRBs are firing. The BPFE mode calls for the use of the ejection seats and the recovery of the crew at sea (except in the case of an on-the-pad ejection). Details for the use of the seats can be found in Volume 1.

2.4.4.3 Capsule Release and Ejection Mode

After the 35 second abort-delay period but before a Trans-Atlantic Abort, the CRE mode is employed. The Earth Return Module contains the only propulsion system capable of "pushing" the crew capsule off the NLS stack and thus causing a capsule release. With the SRBs firing, the ERM cannot generate enough thrust to do this. However, as soon as the thrust-to-weight ratio drops below 2.53 (six seconds before SRB burnout), the ERM will have enough thrust to perform this abort. In the CRE mode, the crew remains in the capsule until conditions are appropriate for ejection. Crew recovery is made at sea. The crew may elect not to remain in the capsule for landing due to uncertainty about the structural integrity of the capsule on impact with the water. Ejection seats will be used in this event.

2.4.4.4 Trans-Atlantic Abort Mode

This is a capsule release abort where the capsule lands at a secondary Space Shuttle landing site in Banjul, The Gambia. (This site is only eight kilometers off of the nominal launch ground track.) Ejection seats may also be used as required. This is very similar to the Space Shuttle TAA mode.

2.4.4.5 Abort-Once-Around Mode

This abort mode is also derived from the Space Shuttle program. If there is enough propellant to do an AOA but not enough for an Abort-To-Orbit (Mode 6) this mode is enacted. Like the Space Shuttle, it calls for a landing at Edwards AFB, in California. Other possible landing sites include White Sands, New Mexico and KSC, Florida.

2.4.4.6 Abort-to-Orbit Mode

Because the launch vehicle and all stages of the payload have single engine-out capabilities, it is possible to reach the nominal 200 km orbit after an engine failure, if all other critical systems are functioning nominally.

2.5 Precursor Mission Launch Events

This section describes the events specifically related to the two NLS launches for the precursor mission.

2.5.1 Launch Window and General Launch Schedule

The precursor mission will require two launches to place the required amount of mass into LEO. The Primary Trans-Lunar Injection stage (PTLI) will be launched in the same manner as for the piloted mission, remaining in a 275 km circular orbit for rendezvous and docking with the precursor payload. The precursor payload will be launched in a similar manner as the piloted payload, with transfer to the 275 km orbit determined to facilitate rendezvous operations.

Since the fuel boil-off rate for the PTLI stage is not a driving factor, the delay between these two launches is flexible. As in the piloted mission, the delay could run from 90 minutes (one orbit) to one month. See Table 2-6 on the following page.

Table 2-6: Precursor Launch Order, Delays, and Considerations

Launch Order	Payload	Delay of Launch	Considerations
1	PTLI stage	(none)	Maintain orbit
2	Precursor	90 min - 1 month	Launch Window

2.5.2 Vehicle Assembly

The processing and assembling process for the precursor payload will closely follow that which is used by the Space Shuttle program at the present time. Figure 2-15 illustrates the Kennedy Space Center (KSC) payload processing flow used for the Space Shuttle. For the precursor payload, the process will be the same as in the figure, with the following exception: after a horizontal checkout of the precursor payload, it will move directly from the Horizontal Payload Processing Facility to the Vehicle Assembly Building (VAB). A vertical checkout option for the precursor payload would require transporting the payload from the Vertical Processing Facility to either the VAB or directly to the launch pad (LC 39 A/B).

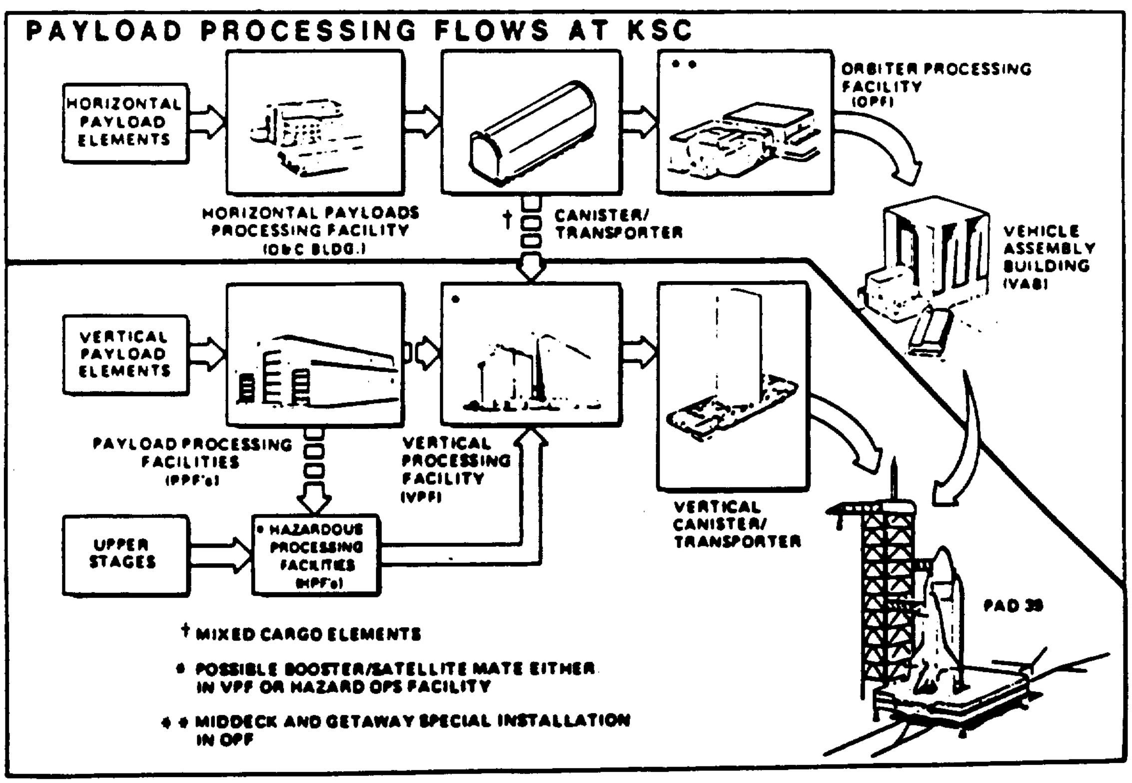


Figure 2-15

The Payload Processing Flows at KSC

The NLS launch vehicle itself will be assembled in the VAB, in the same way as for the piloted mission.

2.5.3 Launch Sequence

The launch sequence for the precursor mission is practically identical to the piloted mission. Refer to Table 2-5 for the detailed launch sequence.

2.5.4 Launch Abort Modes

The abort modes available to the precursor mission consist of a subset of the abort modes for the piloted mission. The two main goals of the precursor mission abort sequences are (1) range safety, and (2) safing of spacecraft components, where possible. If spacecraft components can be preserved and safely placed on the lunar surface or left in Earth orbit for subsequent usage, the abort mode selected will reflect this decision. For all other circumstances, spacecraft components will be destroyed, deorbited, or placed in a benign trajectory to ensure range safety and/or minimize any orbital debris hazard.

The three abort modes available for the launch of the precursor mission are: (1) Redundant-Set-Launch-Sequencer abort, where the countdown is halted before SRB ignition, (2) Abort-to-Orbit, with single engine failure, and (3) destruction of the launch vehicle and its payload by the Range Safety Officer, in case the vehicle's trajectory takes it over populated areas.

2.6 Ascent Trajectory

The NLS capability and trajectory analysis was done using a planar trajectory model over a non-rotating, spherical earth. (Rotational effects were considered by changing the initial conditions to reflect an easterly velocity.) The thrust, component weights, and total vehicle weight was modeled using Shuttle thrust profiles, g limits and a constant fuel flow rates. The analysis assumed a constant pitch rate after clearing the tower at t=6.0 sec. until a pitch of 0° was reached. A coefficient of drag based on Shuttle values was used with a vehicle cross sectional area of 100.8 m^2. Temperature and gravity were assumed to be constant. The equations of motion, as presented in Griffin and French, [Griffin, 1991] are as follows:

$$dV/dt = (T \cos\alpha - D) / m - g \sin\gamma \quad (2\text{-}2)$$

$$V\, d\gamma/dt = (T \sin\alpha + L) / m - (g - V^2 / r) \cos\gamma \quad (2\text{-}3)$$

$$ds/dt = (R/r)\ V \cos\gamma \qquad (2\text{-}4)$$

$$dr/dt = dh/dt = V \sin\gamma \qquad (2\text{-}5)$$

$$D = 1/2\rho V^2 S C_D \qquad (2\text{-}6)$$

where:

V = Inertial velocity	m = mass @ time = t
R = Earth Radius	D = Drag force
h = Height above surface	C_D = Drag force
r = Radius from earth center	ρ = Atmospheric density
s = Down-range travel	S = Drag reference area
γ = Flight path angle	α = Thrust vector angle
T = Thrust @ time = t	

Table 2-7 shows the results of the ascent analysis for a 91 mt payload to be placed in an elliptical orbit with eccentricity of 0.045 at MECO. This orbit will allow the vehicle to coast to its initial orbital altitude of 200 km where the circularization burn will take place. This analysis assumes that the ascent trajectory specified will place the vehicle in the 200 km elliptical orbit. The NLS analysis gives a baseline trajectory from which loading, velocity, and trajectory information can be obtained.

The NLS will follow a similar launch profile to the Shuttle. SRB burnout and staging will occur at 123 sec. Main Engine Cut Off (MECO) will occur at t=416.5 sec, at an altitude of 127 km. The first circularization burn will take place at approximately t=967.5 sec after launch at an altitude of 200 km. Later, at a time determined by ground control, an additional burn sequence is performed, leaving the vehicle in a circular orbit at an altitude of 275 km.

2.6.1 Sequence of Events

Table 2-7 on the next page is a chronology of ascent events for the NLS vehicle. The ascent is similar to the Shuttle's. [Suit, 1992]

Table 2-7: NLS Ascent Event Sequence

Time	Altitude (km)	Event
t = -3 sec	0	Space Transportation Main Engines (STME) ignite
t= 2.64 sec	0	SRB's ignite
t= 3 sec	0	Lift-off
t= 6 sec	0.126	Tower cleared, start constant pitch rate trajectory
t= 45 sec	11.3	STME's throttle back to 75% for max. Q
t= 64 sec	23.4	STME's throttle back to 100%
t= 123 sec	86	SRB burnout
t= 134 sec	98	SRB's jettisoned
t= 401 sec	128.4	STME's throttle back to 75% to remain in g limit
t= 416.5 sec	128.4	MECO, Elliptical orbit with e = 0.045
t= 432 sec	131	Core stage and nose cone jettisoned
t= 967.4 sec	198	Circularization burn starts
t= 1000 sec	200	Circularization burn complete

2.6.2 Altitude, Downrange, and Pitch Profile

Figure 2-16 on the following page shows a plot of launch vehicle altitude vs. downrange distance through MECO. After MECO, the core vehicle burns up over the Indian Ocean while the SRBs are retrieved in the Atlantic. The SRBs free-fall to an altitude of 4.6 km where the nose cone is ejected and the drogue and parachute are pulled out. The SRB's splash down at about a velocity of approximately 88 m/s and a down range distance of 150 km. [Kaplan, 1978].

Figure 2-17 gives the pitch profile of the NLS vehicle. The pitch profile is determined by MECO altitude, vehicle orientation, and weather conditions. In most cases the sole factor driving pitch variation is the wind. Mean wind data is available for each month at KSC. The pitch profile used by the guidance system is a result of these mean winds, the type and size of the payload, and the final vehicle orientation. For this analysis, a constant pitch rate trajectory was used. This trajectory can be modified subject to atmospheric conditions at launch, and to obtain the necessary elliptical orbit at MECO.

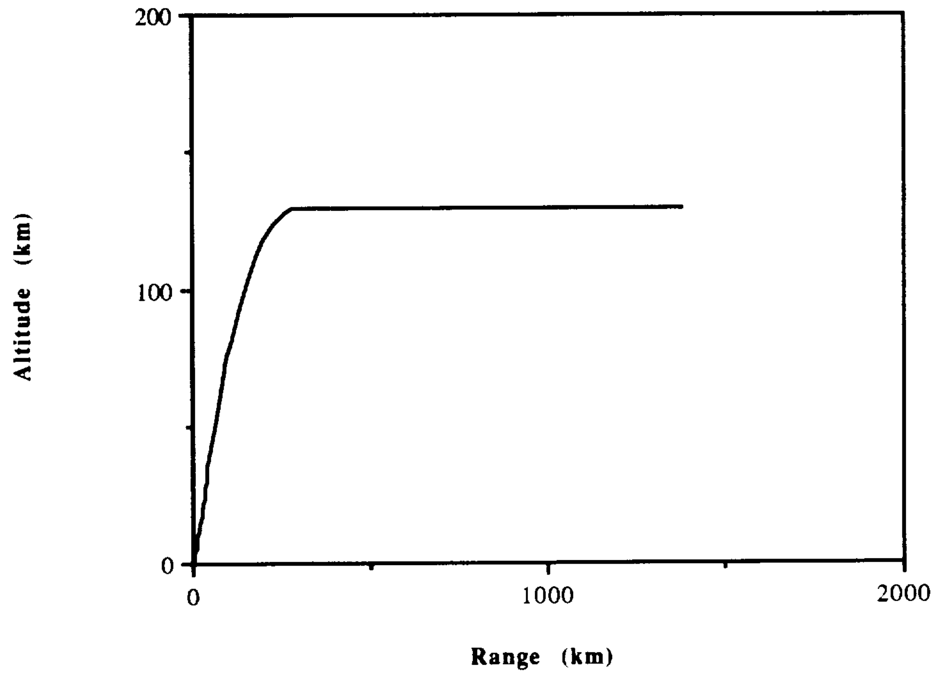


Figure 2-16
NLS Altitude vs. Down Range Trajectory

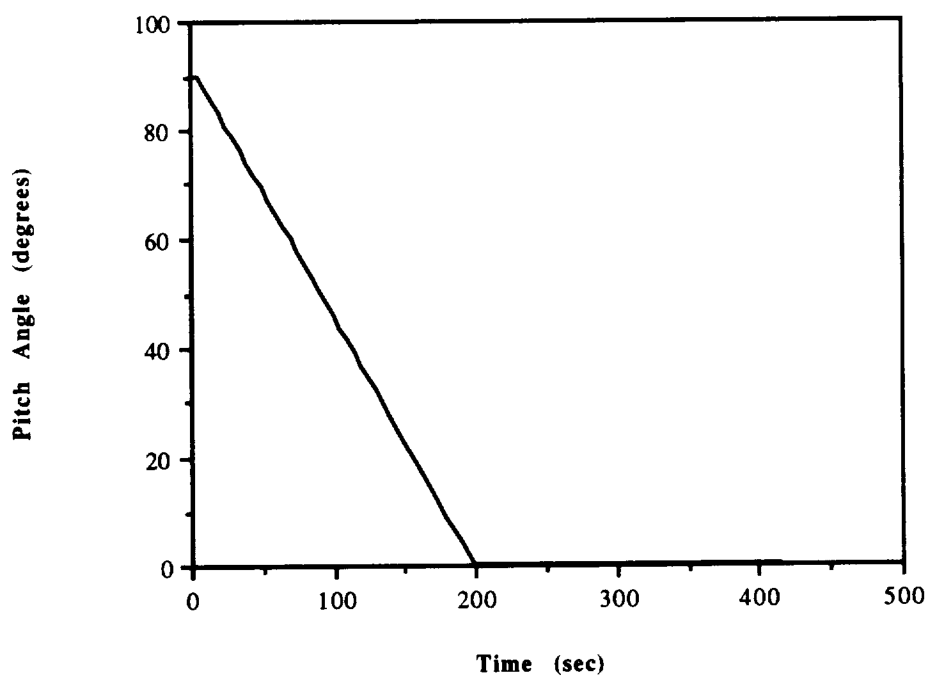


Figure 2-17
NLS Pitch Profile with Time

2.6.3 Modified Pitch Guidance

Most launch vehicles currently employ a standard pitch profile that is derived from the mean winds of the month of launch. Weather balloon measurements hours before launch probe for adverse or unexpected wind conditions. A new system employing a laser Doppler velocimeter will be able to take real time wind measurements as the vehicle is ascending through the winds. This system called Light Detection and Ranging (LIDAR) works similar to a wind shear detection device on an aircraft (Figure 2-18). The laser ranges the winds up to 20 km by measuring the light rays reflected from particles in the air. This wind profile is fed directly into the guidance system of the NLS and allows gimballing of the engines and thrust vectoring for load relief and trajectory optimization. This system is currently being tested simultaneously with Shuttle launches at KSC. [Suit, 1992]

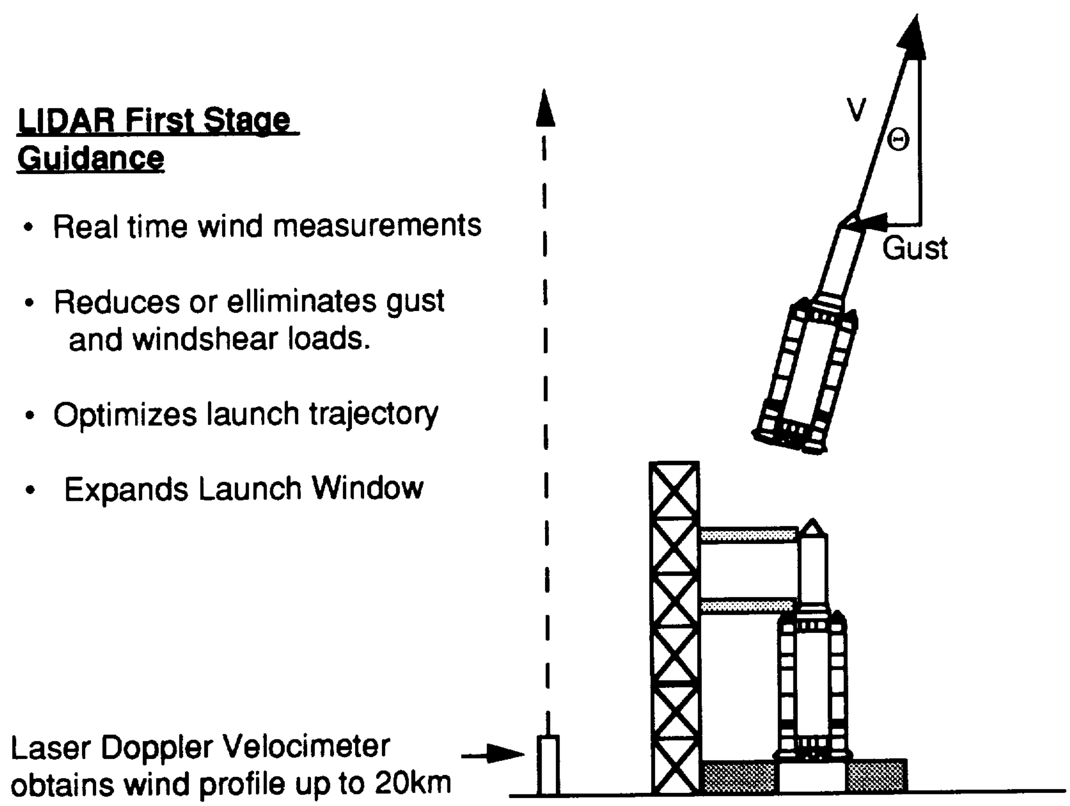


Figure 2-18
LIDAR Modified Guidance for Lateral Load Alleviation and Trajectory Optimization

2.6.4 Orbital Insertion and Circularization

Orbital insertion will be accomplished by using the PTLI stage or the LBM stage, depending on the mission.

At MECO, the vehicle is in an elliptical orbit with an eccentricity of 0.045 with apogee at 200 km altitude. The insertion burn will be performed near apogee, at 967.4 sec into the flight, for 37 sec with the PTLI or 73 sec with the LBM, leaving the vehicle in a final circular orbit of 200 km. Figure 2-19 shows the total trajectory to LEO as well as the SRB and core vehicle trajectories. The SRBs are recoverable approximately 150 km downrange.

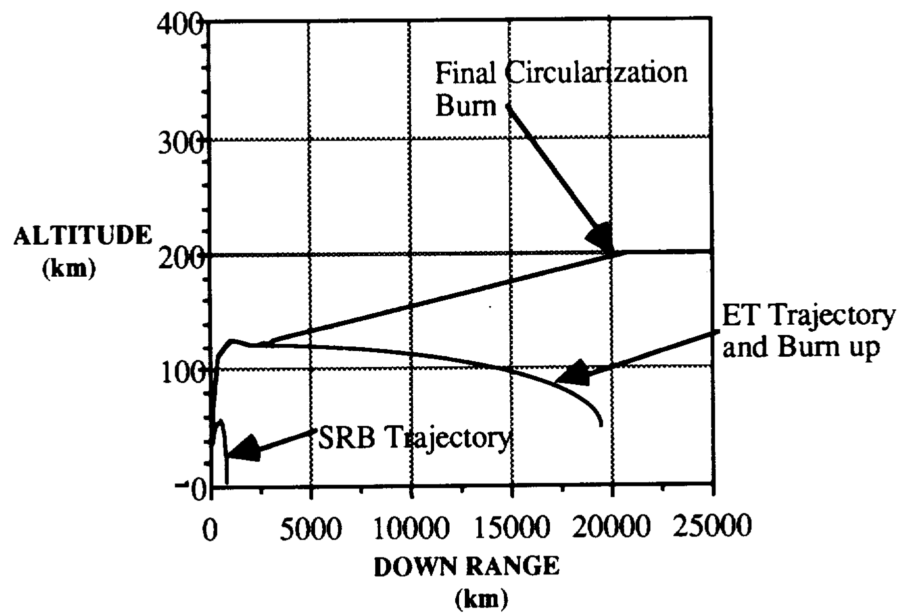


Figure 2-19
Total NLS Trajectory to LEO

2.6.5 Orbital Degradation

Depending on the cross sectional area of a vehicle in orbit, it will experience an altitude degradation similar to that shown in Figure 2-20, derived from Space Shuttle data. For the PTLI stage with a circular cross sectional area of 28.3 m^2 (6 m diameter) oriented along the flight path, it will lose approximately 0.25 km of altitude per day [NASA, 1982]. Rendezvous, docking, and trans-lunar injection should be done within a week so that the PTLI stage will be in the proper orbit. This rendezvous process would normally be completed as quickly as possible but in the case of unforeseen problems it would be beneficial to have the station-keeping capability in reaction control system (RCS) clusters on the PTLI stage without using its propellant. The PTLI fuel burn-off will ultimately determine the maximum allowable orbit stay time before the mission must be scrapped for lack of trans-lunar fuel.

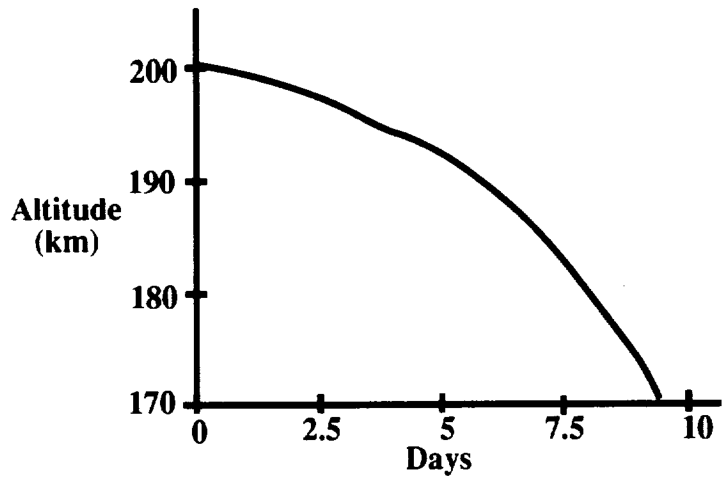


Figure 2-20
TLI Orbital Degradation with Time

2.6.6 Instrumentation Interfaces

The NLS instrumentation system includes hardware for guidance, communication, power, staging, and subsystem notification. All instrumentation systems will be connected to the ground via an umbilical line so that all up testing can be accomplished on the ground. Existing instrumentation packages and units will be integrated on the instrumentation ring of the NLS. Antennas may originate from other areas of the vehicle and redundant features will be utilized as required.

Communications during launches are extremely important. The communications system must provide for the reception of commands and tracking data from and allow the transmission of status data to ground control. During the launch phase, either DSN or TDRSS may be used. The modules of the spacecraft will have stub antennae installed for use in near-earth operations. These low-gain antennae will be used up to LEO.

2.6.7 Vehicle Induced Launch Environment

The vehicle induced launch environment includes thermal deviations, acoustical vibrations, and acceleration loading.

The launch environment will only raise the temperature of the payload about 2.5° C. The vehicle will have no thermal control options besides passive insulation.

During powered ascent, engine exhaust noise and aerodynamically generated vibration will subject the vehicle to random perturbations while in the atmosphere. These fluctuating pressures cause structural vibrations dependent on the load transfer members connected to the payload. No active vibration damping or relief is available on the vehicle, so the payload must be designed to withstand these fluctuations [NASA, 1982]. Shuttle test data is plotted in figure 2-21.

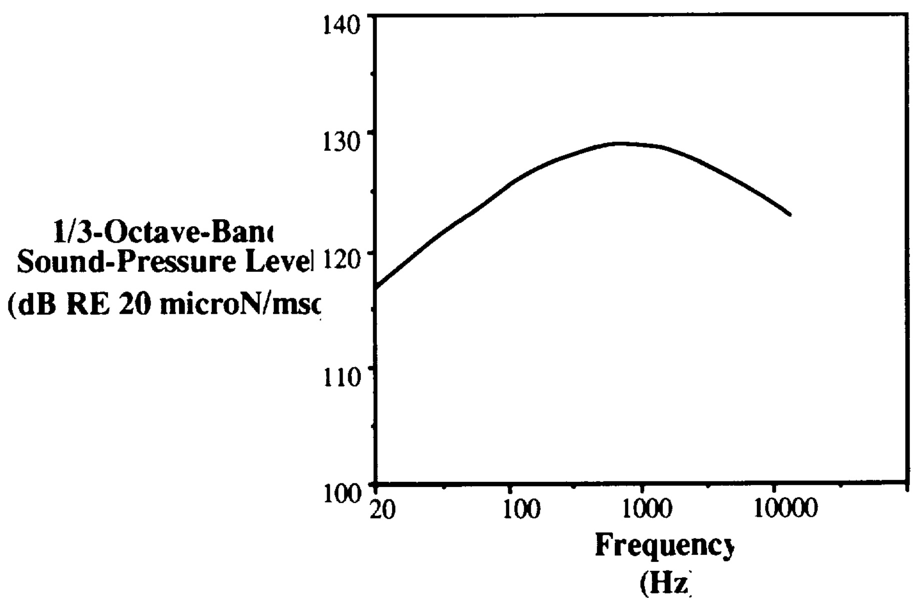


Figure 2-21
Launch Induced Acoustical Vibrations

Figure 2-22 shows the thrust-to-weight profile, and thus the loading on the NLS launch vehicle. The maximum axial loading is approximately 3.8 g's and the maximum lateral loading is approximately 2.3 g's. Structures will be designed with a factor of safety of 1.4.

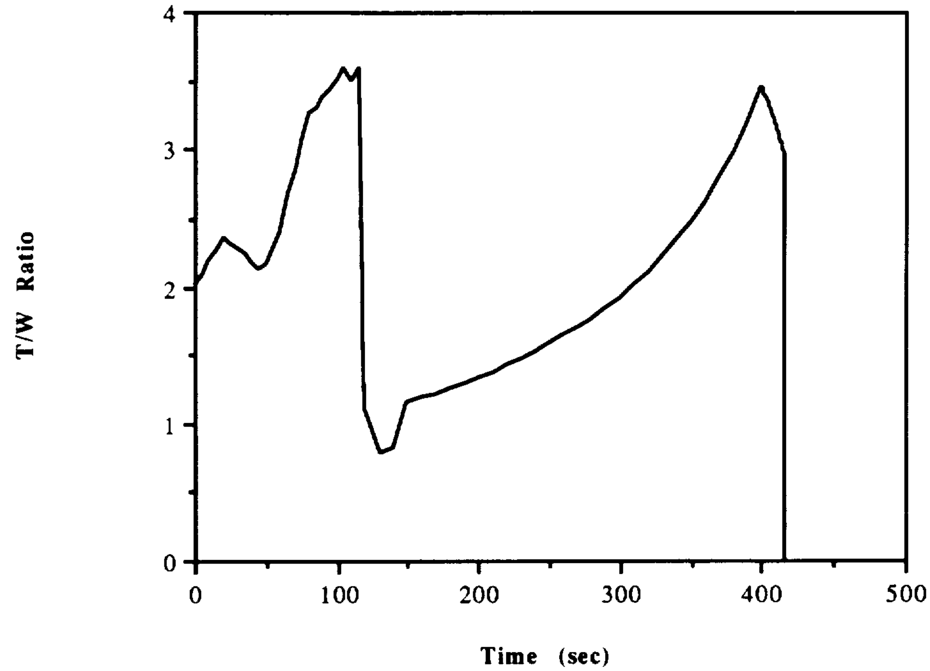


Figure 2-22
NLS Thrust to Weight Ratio

3. Primary Trans-Lunar Injection Stage

The mission of the Primary Trans-Lunar Injection (PTLI) stage is to place its payload on a trajectory toward the Moon. The stage provides the initial boost from low Earth orbit for both the precursor and the piloted lunar missions. Mission analysis shows that the propellant mass required for the entire Trans-Lunar Injection burn outweighs the lift capability of the launch vehicle. Therefore, the PTLI stage provides the initial part of the trans-lunar injection burn, and the Lunar Braking Module (LBM) completes the burn. Since this stage cannot provide the entire ΔV needed, it is called the 'Primary' Trans-Lunar Injection stage.

3.1 Stage Requirements and Operations

This section describes the mission requirements and profile for the Primary Trans-Lunar Injection stage.

3.1.1 Requirements

For mission success, the PTLI stage must meet several requirements. The driver in the PTLI stage design is the launch vehicle capacity. A systems level analysis taking into consideration the launch vehicle limitations produced the following requirements for the PTLI stage:

- To deliver 2530 m/sec ΔV for a 90 metric ton payload.
- To stay within the loading limits.
- To dock with the payload in low Earth orbit.
- To maintain orbit for two launch windows (one month).
- To provide the initial part of the orbital insertion burn.

Within the scope of these basic requirements, the PTLI stage maximizes the propellant capacity to orbit.

3.1.2 Budgets

This section details breakdowns in mass, power, and propellant allocations for the PTLI stage.

3.1.2.1 ΔV Budget

The PTLI stage must perform five major independent burns and several small stationkeeping burns. Table 3-1 on the next page lists the ΔV's for the different burns.

Table 3-1: PTLI operations with ΔV's, Burn Times, and Propellant Usage

Operation	ΔV	Engines	Burn Times	Fuel Used
Orbital Insertion	177 m/s	Main	36.6 sec	3,848 kg
Orbit Change (apogee)	22.04 m/s	RCS	1048 sec	673.3 kg
(perigee)	21.98 m/s	RCS	1038 sec	666.7 kg
Drag recovery	1.21 m/s	RCS	244.6 sec	39.3 kg
Orbit Keeping	25 m/s	RCS	variable	810 kg
Trans-Lunar Injection	2,530 m/s	Main	759.5 sec	79,745 kg

3.1.2.2 Propellant Budget and Storage

Propellant calculations used the ideal rocket equation and took into account the mission ΔV's listed in Table 3-1, the engine characteristics listed in Table 3-2, and several other factors that increase the necessary propellant volume. The main propellant includes an extra 1.65% of propellant. This figure includes a projected boiloff of 0.20% over the one month orbital stay and an extra 1.45% propellant that will be unusable because the propellant cannot be completely drawn out of the tanks. The RCS propellant has only a 1.5% margin for unusable fuel. Table 3-3 lists several propellant characteristics used for the tank design.

Table 3-2: PTLI Engine Characteristics

Engine Name	Number	Specific Impulse	Mass Flow
RL10A-4	5	449 sec	105 kg/sec
R4-D	16	312 sec	0.1606 kg/sec/engine

Table 3-3: PTLI Propellant Characteristics

Propellant	Mass	Density	Volume	Pressure
Liquid Oxygen	71,924 kg	1230 kg/m^3	61.40 m^3	296 kPa
Liquid Hydrogen	13,076 kg	71.0 kg/m^3	193.39 m^3	197 kPa
MMH (hydrazine)	844 kg	878.8 kg/m^3	1.0 m^3	1516 kPa
Nitrogen Tetroxide	1,391 kg	1447 kg/m^3	1.0 m^3	1516 kPa

Since it is storable at room temperature, the reaction control engine propellant will be loaded during launch preparations several days before the actual launch. The cryogenic propellant will be added just prior to liftoff, as the propellant will begin to boil off immediately. Due to the extensive insulation on the cryogenic tanks, however, only a small percentage of the propellant will evaporate while the vehicle sits on the pad.

3.1.2.3 Mass Budget

The launch vehicle capacity sets the mass requirements on the PTLI. The launch vehicle places 100 metric tons onto an orbital trajectory. The PTLI stage must meet this 100 ton limit while meeting the mission requirements stated above. Table 3-4 provides a system level breakdown of the masses on the PTLI stage both at liftoff and before the Trans-Lunar Injection burn.

Table 3-4: PTLI Mass breakdown

(a) At Launch		(b) Before Trans-Lunar Injection After Orbital Insertion	
Subsystem	Mass	Subsystem	Mass
Structures	6,550 kg	Structures	6,550 kg
Communications	149 kg	Communications	149 kg
Guidance	28 kg	Guidance	28 kg
Power	519 kg	Power	519 kg
Insulation	1,662 kg	Insulation	1,662 kg
Propulsion	1,550 kg	Propulsion	1,550 kg
Propellant (Main)	85,000 kg	Propellant (Main)	81,152 kg
(RCS)	2,235 kg	(RCS)	2,235 kg
Nose Cone	820 kg		
Total	98,513 kg	Total	93,845 kg

3.1.2.4 Power Budget

Several subsystems require power during flight. As the PTLI stage is autonomous for a significant portion of its designed operability, the PTLI stage contains its own power source for all systems. Guidance, communications and control equipment uses 978 W of power continuously during the entire 40 day lifetime of the PTLI stage. Engine burns require large amounts of power over short periods of time to actuate valves and provide ignition. Main engine ignition uses 1,905 W of power for startup and only 1,500 W of power to shut down. Each RCS engine needs 50 W of power over the duration of the operation. The current power design allows for 6 main engine operations, although only two are planned. Power is available to operate 4 RCS thrusters at a time for a total burn of 3,500 sec. Figure 3-1 displays a nominal power versus time curve for the entire 40 day flight. The power spikes for the main engine operation last only a few seconds.

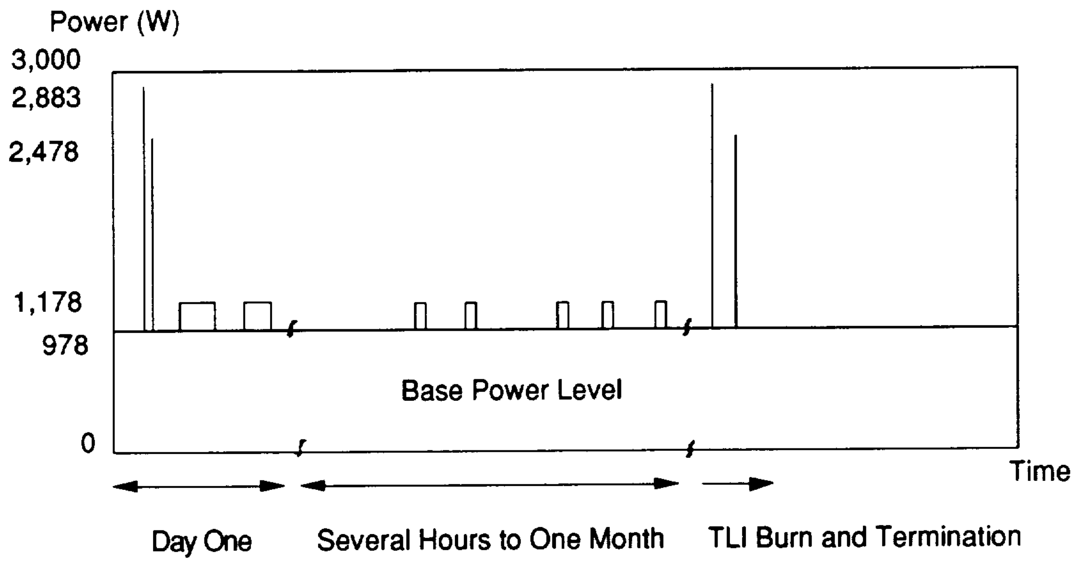


Figure 3-1
Power Required by PTLI vs. Time

3.1.3 *Mission Profile*

3.1.3.1 Launch

During the launch phase, the PTLI stage performs several maneuvers. Precisely 432 sec after lift-off, several seconds after engine cut-off, the launch vehicle will separate from the PTLI stage using an explosive ring. Seconds after the launch vehicle separates, the PTLI stage also jettisons the nose cone and the four shrouds covering the RCS jets. Upon reaching apogee, the PTLI stage fires its five main engines for orbital insertion. The orbital insertion burn last 36.6 seconds and provides 177 m/s Δv. The orbital insertion burn uses 3,848 kg of main propellant. The onboard guidance system and ground tracking will determine the actual orbit of the PTLI stage after orbit is achieved.

3.1.3.2 Orbital Operations

After reaching orbit, the PTLI stage performs several small orbit changing maneuvers. After establishing its orbit, the PTLI stage uses four of its RCS engines to raise its orbit from 200 km to 275 km altitude. This orbit change reduces the altitude loss due to drag on the stage from 13 km per month to 2 km per month. The transfer orbit is a Hohmann ellipse, requiring 22.04 m/sec Δv for the perigee burn, and 21.98 m/sec Δv at apogee. The stage will orbit at 275 km altitude until the payload arrives. In the event that the payload takes one month, the PTLI stage requires an additional 1.21 m/sec Δv from the RCS system to return to the 275 km orbit.

The PTLI stage will reach orbit several hours to several days before either the precursor payload or the staffed vehicle. During its orbital stay, the PTLI experiences a gravity gradient and several other sources of drag. These forces, although small, will pull the PTLI stage out of its preassigned orientation and orbit. To maintain the proper attitude, the PTLI stage will use its RCS system when necessary. The RCS system provides up to 25 m/sec Δv for attitude control.

3.1.3.3 Rendezvous

The PTLI stage remains stationary during docking. No propellant is budgeted for the rendezvous. Nevertheless some of the orbit keeping fuel might be available if necessary. After completing the rendezvous, it is necessary to confirm the interfaces between the PTLI stage and the LBM stage.

3.1.3.4 Primary Trans-Lunar Injection Burn

After docking and orientating the vehicle in the right direction, the PTLI stage will begin the Trans-Lunar Injection burn. All five main engines will fire for 759.5 sec, using 80 tons of the main propellant. Either at the end of the 759.5 sec or in the event of multiple engine failures (the stage has an engine out capability), a command will turn off the main engines. Propellant remaining in the tanks at this time will propel the stage during its termination flight.

3.1.3.5 Stage Termination

After separation from the LBM, the PTLI is on an elliptical orbit that returns it to the Earth. Firing the RCS system against the velocity vector at apogee should land the stage in the Earth's atmosphere. Additionally, when the PTLI stage regains communications with Earth, ground controllers will track the vehicle and guide it into the Earth's atmosphere. All remaining propellant can be used for the necessary course corrections. If the stage does not completely burn in the atmosphere, the fragments will enter the Pacific Ocean.

3.1.4 Abort Options

Abort options involving the PTLI stage are almost identical between the piloted mission and the precursor mission.

3.1.4.1 Earth Orbit Abort

During the piloted mission, a return-to-earth abort mode exists any time prior to the Primary Trans-Lunar Injection stage burn. This abort is accomplished using the propulsion system of the Earth Return Module; the PTLI stage is then deorbited using its own primary or secondary propulsion system.

For the precursor mission, the PTLI stage may be deorbited using the primary or secondary propulsion systems in the event of a decision to abort the precursor mission and accept the loss of the propulsive stage. The PTLI possesses a nominal 48-day on-orbit stay duration. Remaining elements of the spacecraft stack may be deorbited or left in orbit as dictated by safety criteria or mission requirements.

3.1.4.2 Trans-Lunar Injection Abort

The Primary Trans-Lunar Injection stage possesses a single engine-out capability (4 out of 5 engines operable) for the entire length of the PTLI burn.

3.2 Stage Design

3.2.1 Stage Configuration

The PTLI stage is configured like any classic rocket. Figure 3-2 shows a general layout of the PTLI and its subsystems.

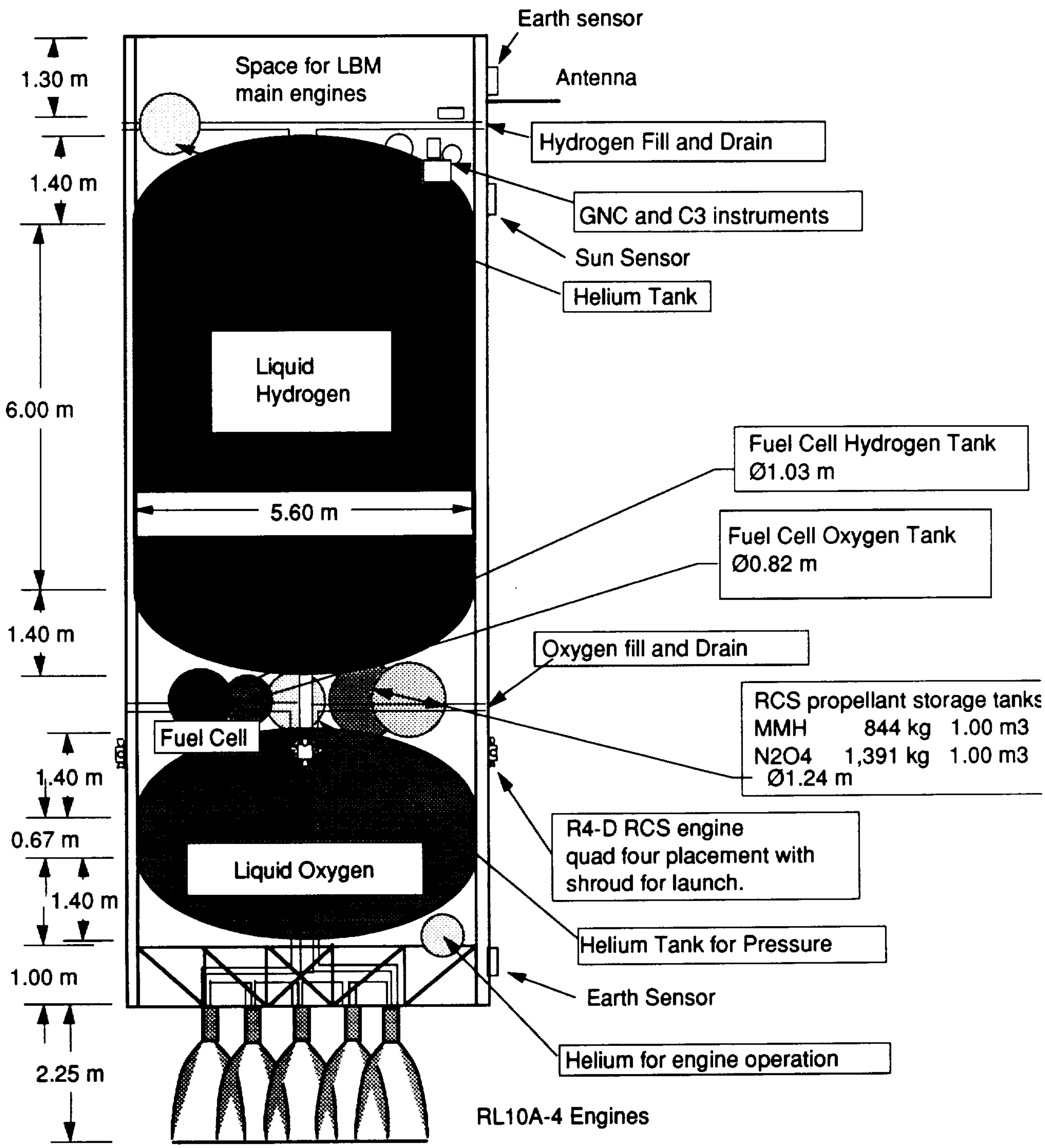


Figure 3-2
Schematic of the PTLI

3.2.1.1 Propulsive System

3.2.1.1.1 Main Engines

The main engine system consists of five RL10A-4 engines, a liquid oxygen tank, a liquid hydrogen tank, helium pressurant gas, and piping. Four RL10A-4 engines are mounted in a symmetric on the bottom of the stage in a square pattern, with the fifth engine placed in the center of the square and the stage. The engines mount directly to a truss, 1.5 m apart to allow for the 4° gimballing of the nozzle cones as shown in Figure 3-3. The five RL10A-4 engines are mounted 1.5 meters apart. The liquid oxygen tank is mounted directly above the truss, followed by the liquid hydrogen tank. The tanks are filled on the launch pad via fill/drain systems above each tank. A helium supply above each tank pressurizes the tank prior to operation.

If a main engine fails, the thrust is kept symmetric by shutting off the opposite engine. If the center engine fails, no additional engines are shut down. Symmetric thrust insures that no torques are induced on the PTLI. Gimbaling cannot align the thrust through the center of mass until some propellant has been expended. Therefore, the shutdown option must be used instead of gimballing to account for engine failure.

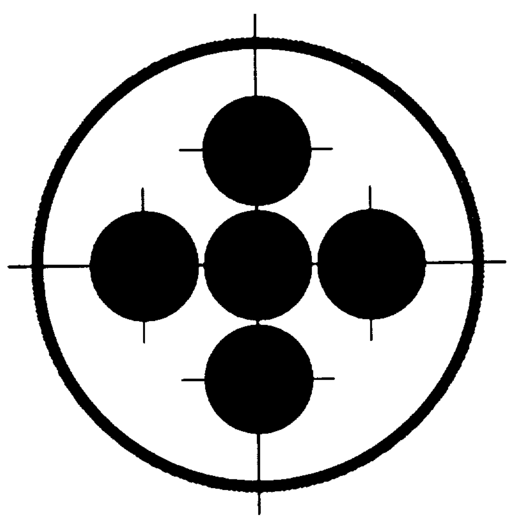


Figure 3-3
Bottom view of PTLI Stage

3.2.1.1.2 RCS Engines

The RCS system contains 16 R4-D engines with propellant tanks and piping. The RCS engines are mounted in groups of four on the sides of the vehicle 90° apart. One engine points in each main direction tangential to the stage. The groups are mounted just below the top level of the oxygen tank. This placement aligns the RCS jets with the stage's center of mass. The monomethylhydrazine (MMH) and the nitrogen tetroxide are loaded into the tanks several days prior to launch. The RCS propellant tanks are situated between the hydrogen and oxygen tanks on one side of the stage. Figure 3-4 gives a cross-sectional view of the area between the liquid oxygen tank and the liquid hydrogen tank, showing the placement of the RCS system and the fuel cells. Piping runs in both directions around the stage allowing for a single level of redundancy.

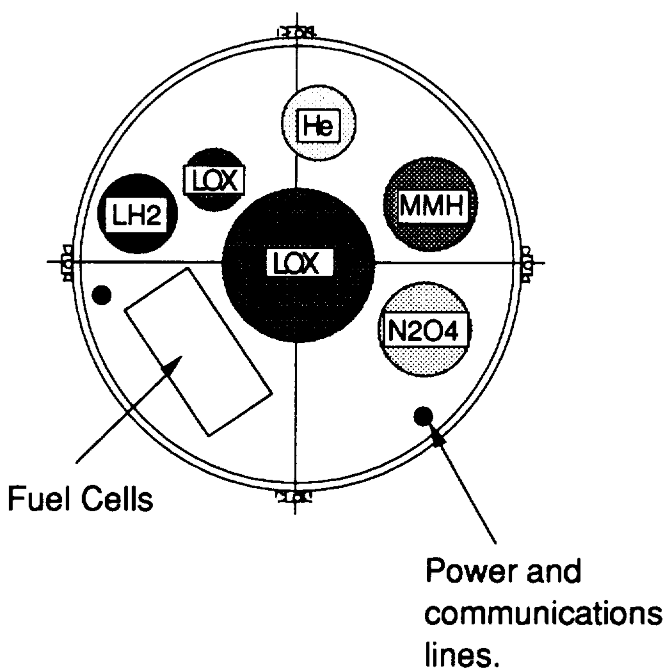


Figure 3-4
Cross-section showing RCS placement

3.2.1.2 Power

The power subsystem consists of secondary liquid hydrogen and liquid oxygen tanks, a fuel cell and a distribution bus. The tanks and the fuel cell, like the RCS systems, sit between the main oxygen and hydrogen tanks (see Figure 3-4). Two transmission lines, 135° apart, provide for a singly redundancy.

3.2.1.3 Structures

The structural subsystem consists of an engine support truss, tank supports and the exterior casing. The support truss is one meter in length, allowing for placement of the main propellant piping inside the truss. Supports from the side walls hold the tanks in place. Since the supports add a load to the structure at the point of attachment equal to the mass of the tank, structures placed the oxygen tank below the hydrogen tank.

3.2.1.4 GNC and Communications

The control components reside either above the hydrogen tank or in a more appropriate place depending on the mission. Earth sensors are placed on the same side of both ends of the PTLI stage. A sun sensor is mounted along with the top Earth sensor, along with one of the eight antennas for the PTLI stage. The other antennas ring the circumference of the stage, providing for communications with the stage regardless of orientation.

3.2.2 Vehicle Interfaces

This section documents the interfaces for the PTLI.

3.2.2.1 Launch Vehicle

Figure 3-5 shows how the PTLI connects to the launch vehicle.

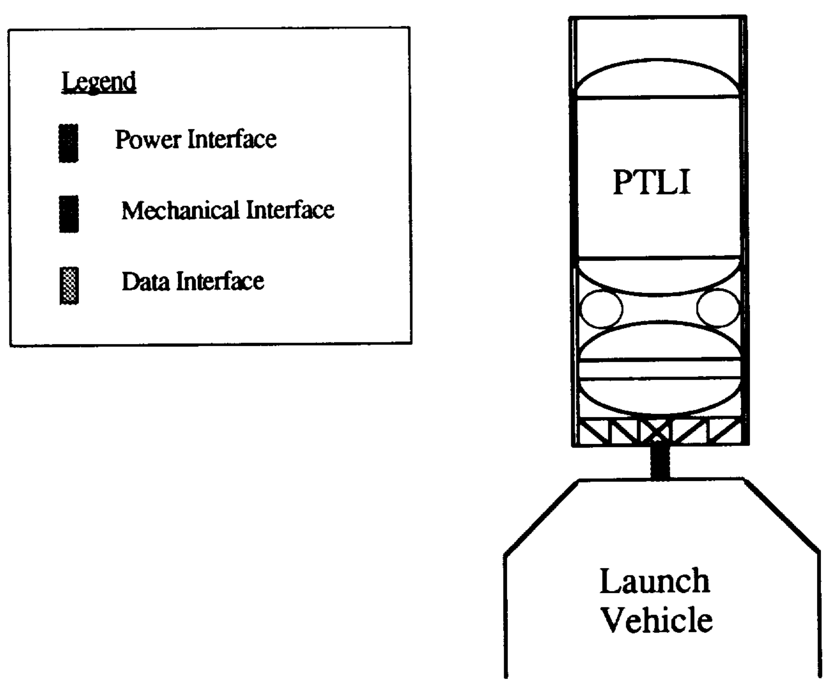


Figure 3-5
PTLI Interface with Launch Vehicle

The mechanical interface between the PTLI and the launch vehicle consists of an explosive ring which is attached to the top of the truss located at the lower end of the PTLI.

3.2.2.2 Nose Cone

Figure 3-6 shows how the PTLI connects to the launch nose cone.

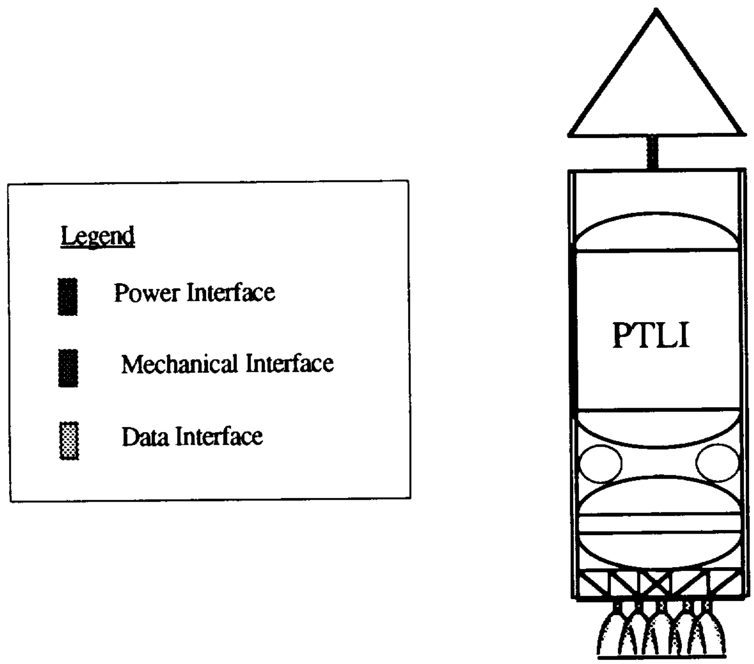


Figure 3-6
PTLI/Nose Cone Interface

The mechanical interface between the PTLI and the nose cone consists of explosive bolts.

3.2.2.3 Lunar Braking Module

Figure 3-7 on the next page shows how the PTLI connects to the Lunar Braking Module.

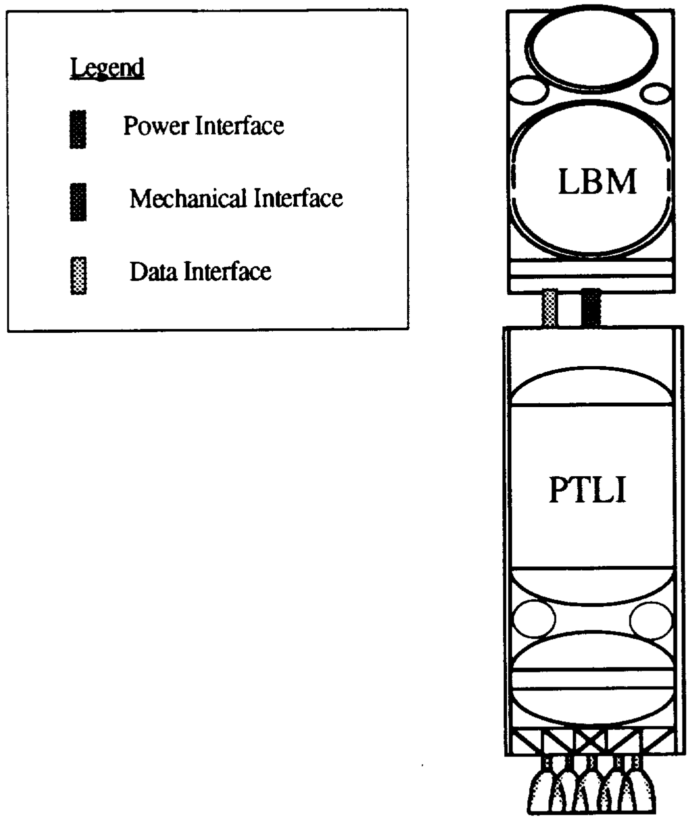


Figure 3-7
PTLI/LBM Interface

3.2.2.3.1 Mechanical Interface

The mechanical interface between the PTLI and LBM consists of docking latches for rendezvous and docking. For stage separation, the interface has explosive bolts between the PTLI and LBM.

3.2.2.3.2 Data Interface

The data interface betweem the PTLI and LBM will transmit information to the CM which monitors the status of the tanks and engines.

3.2.2.4 PTLI Stage Docking Latch System

The design of the docking latch system shown in Figure 3-8 is based on a scaled-up version of a design proposed for shuttle docking with the space station.

The docking system consists primarily of four pairs of latches spaced equally around the circumference of each PTLI stage. During the docking maneuver, four-inch long, fin-like guides slide into slots in the upper stage (piloted or habitat cluster). These guides can accommodate lateral misalignments of 8 cm and roll/pitch/yaw misalignments of 5 degrees, reducing the requirements of the RCS thruster system. A mass of 50 kg per patch or 400 kg per PTLI assembly is estimated.

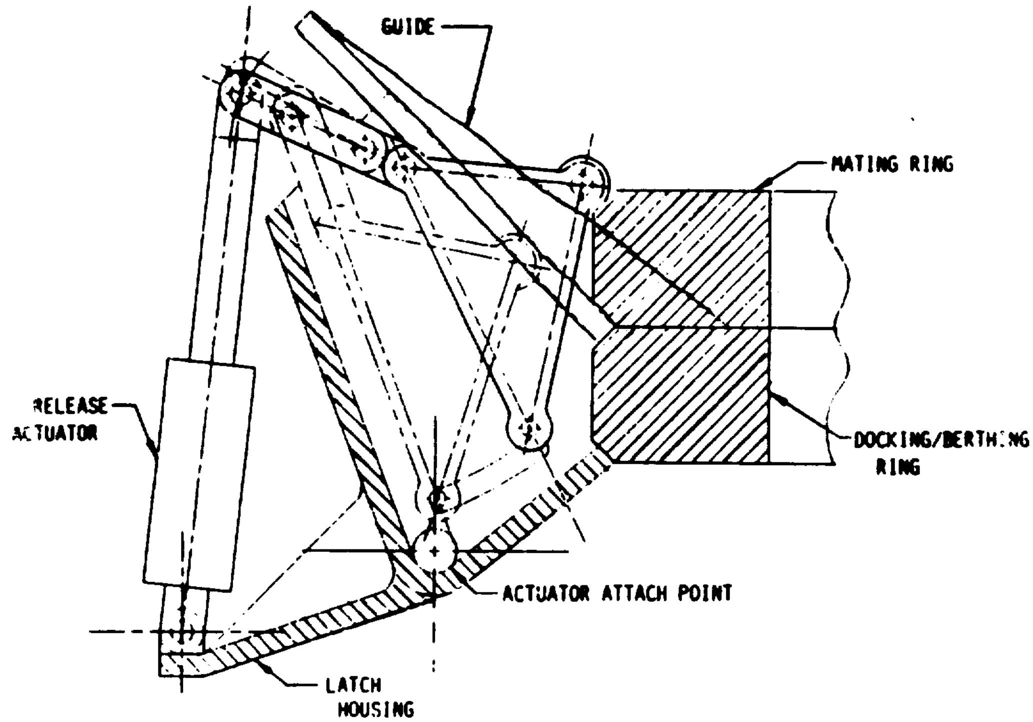


Figure 3-8
Diagram of Docking Latch

After the initial alignment, electromechanical actuators perform a 30 cm stroke to clamp the mating and docking rings together. During docking, each latch is subjected to 500 N of equivalent axial loading. During the translunar injection maneuver, the axial loads are carried by the hull of the PTLI stage.

3.3 Subsystem Design

This section gives greater depth about subsystems on the PTLI.

3.3.1. Structural Design

A summary of the PTLI structural design is included here as a service to the reader. For those interested in the methods of the design or desire to understand the structural trade-off that were involved in the choice of this structural configuration you are referred to section 2.1.4 and 2.2.1 in Volume II.

The geometrical description and masses are summarized in Tables 3-5, 3-6, and 3-7.

Table 3-5: PTLI Hydrogen Tank Design

Hydrogen Tank	
Hydrogen Mass	13076.92
Hydrogen Volume	184.18
Hydrogen Tank Volume	193.39
Hydrogen Tank Radius	2.80
Hydrogen Tank Cap Radius	1.40
Hydrogen Tank Cap Volume	45.98
Hydrogen Tank Main Volume	147.41
Hydrogen Tank Main Height	5.99
Hydrogen Tank Cap Eccentricity	0.87
Hydrogen Tank Cap Area	57.39
Hydrogen Tank Body Area	105.30
Hydrogen Tank Area	162.69
Hydrogen Tank Wall Thickness	0.0011
Hydrogen Tank Structure Mass	257.12
Hydrogen Tank Coating Thickness	0.0010
Hydrogen Tank Coating Mass	1236.44
Hydrogen Tank Height	8.79
Hydrogen Tank Insulation Mass	1757.05
Hydrogen Tank Mass	1493.56

Table 3-6 : PTLI Oxygen Tank Design

Oxygen Tank	
Oxygen Mass	71923.08
Oxygen Volume	58.47
Oxygen Tank Volume	61.40
Oxygen Tank Radius	2.80
Oxygen Tank Cap Radius	1.40
Oxygen Tank Cap Volume	45.98
Oxygen Tank Main Volume	15.42
Oxygen Tank Main Height	0.63
Oxygen Tank Cap Eccentricity	0.87
Oxygen Tank Cap Area	57.39
Oxygen Tank Body Area	11.02
Oxygen Tank Area	68.41
Oxygen Tank Wall Thickness	0.0014
Oxygen Tank Structure Mass	144.92
Oxygen Tank Coating Thickness	0.0010
Oxygen Tank Coating Mass	519.91
Oxygen Tank Height	3.43
Oxygen Tank Insulation Mass	738.81
Oxygen Tank Mass	664.82

Table 3-7: PTLI Configuration Summary

Configuration	
Stage Radius	3
Total Height	16.46
Insulation Mass	2496
Casing Mass	3835
Rocket Truss Mass	557
Tank Mass	2158
Structural Mass	6550
Engine Mass	835
Stage Dry Mass	9881
Stage Wet Mass	94881
Vehicle Wet Mass	181211
Structural Mass Fraction	7%
Structural Fuel Fraction	7.7%

3.3.2 Propulsion

3.3.2.1 Primary Propulsion System

The primary propulsion system of PTLI Stage is shown in Figure 3-9 on the next page. It consists of five RL10A-4 engines rated at 92,518 N nominal thrust and operating each at a 5.5:1 mixture ratio of oxidizer to fuel. The net positive suction head (NPSH) required by the engine turbopumps is provided by pressurizing the vehicle propellant tanks with helium gas stored at 272 atm. Propellants are delivered to the main engine turbopumps through feed ducts from the vehicle propellant tanks. The feed ducts contain flex joints to accomodate engine gimbaling and are overwrapped with a three-layer, double aluminized Kapton radiation shield.

The primary propulsion engines run on a bipropellant combination of liquid oxygen oxidizer and liquid hydrogen fuel. Both propellant tanks are cylindrical with semi-spherical endcaps, and are constructed of a thin steel core overwrapped with pre-stressed graphite composite fibers and a 20 cm layer of aluminized Kapton insulation. The oxidizer tank is 3.43 m tall and 5.6 m in diameter; the fuel tank is 8.79 m tall and 5.6 m in diameter.

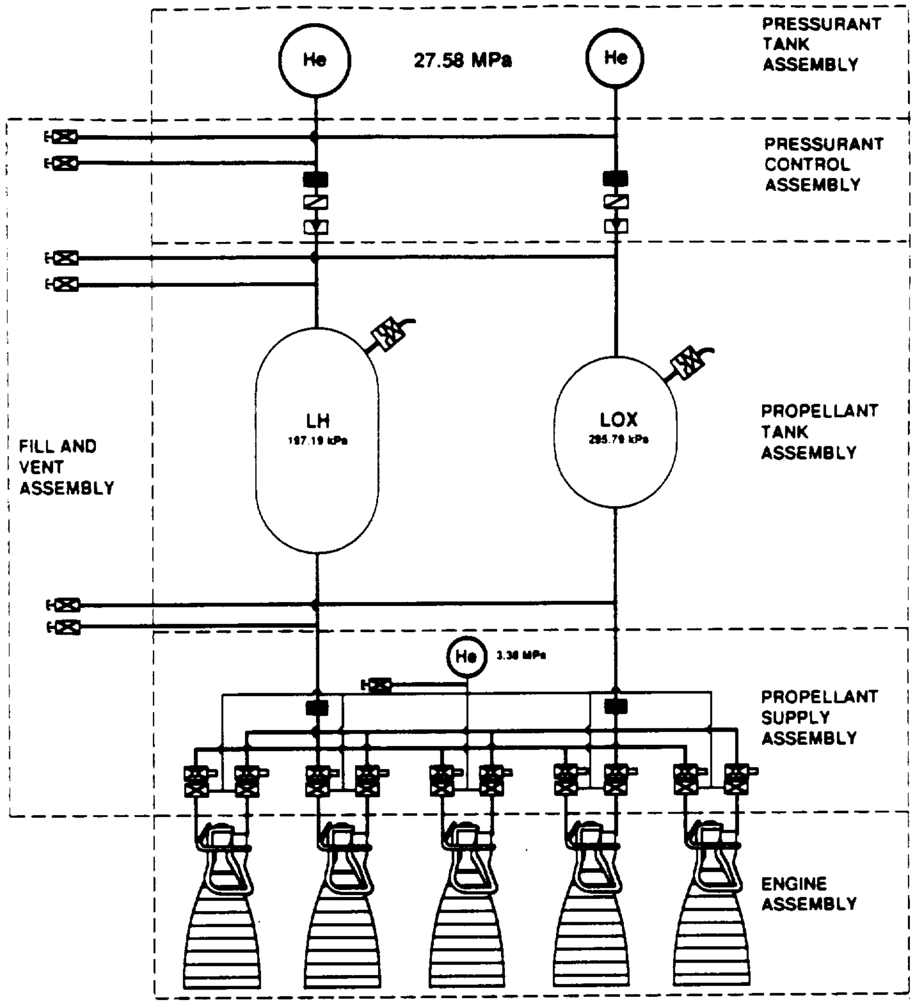


Figure 3-9
PTLI Primary Propulsion System

Pneumatically actuated prevalves located at the propellant tank outlets provide series redundant backup for the engine inlet shutof valves. A parallel set of pyro valves and solenoid valves upstream of the pneumatic actuation control solenoid valves provides two-failure tolerance against inadvertent opening of the engine inlet shutoff valves. The pyro valves will be fired open after the PTLI stage is deployed a safe distance from the launch vehicle upper stage. The system also has manual fill and drain valves to load propellant and pressurant gas into the system, as well as additional manual valves for system leak checking on both sides of the pyro-isolation valves and regulators. Check valves insure that the fuel and oxidizer can never mix anywhere in the system, except in the engine. Finally, pressure transducers, filters, temperature sensors, and line and component heaters are provided to ensure proper subsystem operation. A mass distribution of the entire propulsion system is given in Table 3-8.

Table 3-8: Mass Distribution of PTLI Primary Propulsion System

COMPONENT	MASS [kg]
Empty Fuel Tank	1,493
Fuel Mass	12,763
Empty Oxidizer Tanks	664
Oxidizer Mass	70,474
Empty Helium Tanks	164
Helium Mass	152
Monitoring equipment	20 (estimated)
Propellant lines	26 (estimated)
Valves	42
Engine mass (5 RL10A-4 engines)	840
TOTAL FUELED WEIGHT	86,638 kg

3.3.2.2 Reaction Control System

The reaction control system of the PTLI stage consists of two redundant subsystems configured as shown in Figure 3-10 on the next page. Each subsystem consists of 8 R-4D thrusters operating on a 1.65 mixture ratio of oxidizer to fuel and fed by two propellant tanks. The thrusters are divided into quadruple clusters which are placed along the periphery of the spacecraft, making a total of 16 thrusters and four propellant tanks for the complete system.

The system utilizes a bipropellant combination of nitrogen tetroxide oxidizer and monometylhydrazine fuel. The propellants are stored in separate spherical tanks of identical size; each tank is 0.76 m in diameter. Both tanks are constructed of a thin steel core overwrapped with pre-stressed graphite composite fibers; no thermal insulation material is required. Propellants are equipped with a Teflon diaphragm positive expulsion device which insures efficient tank evacuation.

A pressurant tank stores helium at about 272 atm, and a quad redundant regulator coupled with a burst disk and relief valve regulates flow. Together, they insure a 15 atm feed pressure to the propellant tanks, even after any single regulator failure. There are burst disks and pyrotechnically actuated squib valves to isolate propellants from the engine (and high pressure gas from the propellant tanks) until the system is ready for operation. This system also has manual fill and drain valves to load propellant and pressurant gas into the

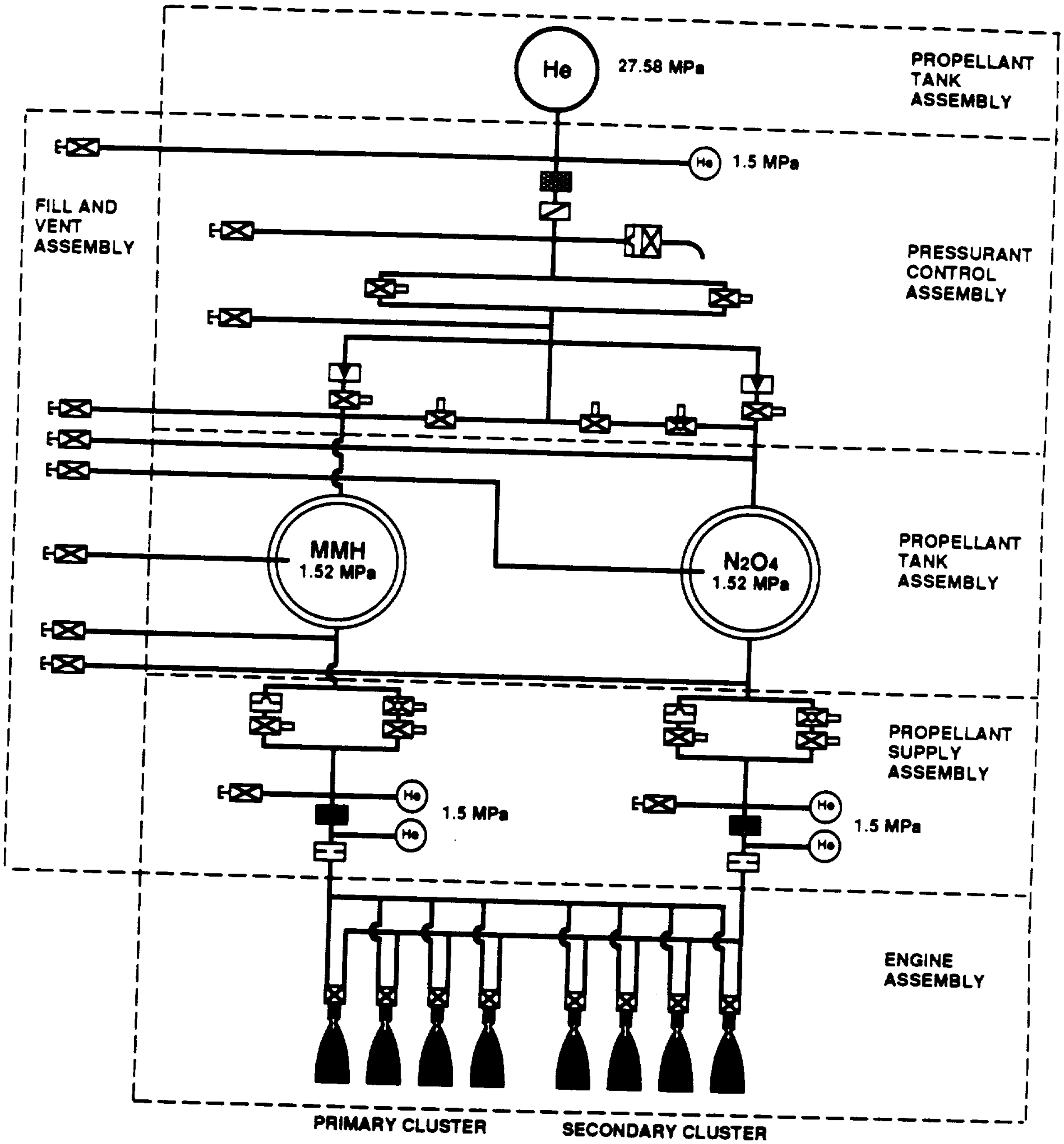


Figure 3-10
PTLI Secondary Propulsion System

system, as well as additional manual valves for system leak checking on both sides of the pyro-isolation valves and regulators. Check valves insure that the fuel and oxidizer can never mix anywhere in the system, except in the engine. Finally, pressure transducers, filters, temperature sensors, and line and component heaters are provided to ensure proper subsystem operation. A mass distribution of reaction control system components is given in Table 3-9.

Table 3-9: Mass Distribution of PTLI Secondary Propulsion System

COMPONENT	MASS [kg]
Empty Fuel Tanks	20
Fuel Mass	192
Empty Oxidizer Tanks	20
Oxidizer Mass	317
Empty Helium Tanks	6
Helium Mass	2
Monitoring equipment	20 (estimated)
Propellant lines	26 (estimated)
Valves	62
Engine mass (16 R4-D engines)	60
TOTAL FUELED WEIGHT	725 kg

3.3.3 Power and Thermal Control

This section describes some energy considerations in the PTLI stage.

3.3.3.1 PTLI Power Supply

The PTLI stage has many of the power needs which the other propulsion stages share in common. It needs electrical power for engine gimbal actuators, engine valves, sensors, explosive bolts, RCS startup, RL-10 ignition, communications, and GNC. As was mentioned the PTLI receives its own on-board power supply. This power supply consists of the fuel cells of choice, alkaline cells (see SLURPP fuel cell trade study) and the reactants for the fuel cells, LOX and LH_2, which will be placed in the separate cryogenic reactant storage tanks of the stage.

It has been estimated that the above power needs sum to about 978W continuous for the performance time duration of the stage, 40 days. Thus the net energy requirement for the power plant is 938.88 kW-hrs. This can be supplied with 368.7 kg of reactants and 19.6 kg worth of fuel cell hardware. The reactants break down as 327.7 kg O_2 and 40.96 kg H_2, or .287 m^3 of O_2 and .577 m^3 of H_2. The volume of the necessary fuel cell apparatus is estimated to be .0244 m^3.

The PTLI has a pair of spherical fuel cell reactant tanks which are separate from the propellant storage tanks. The reactants are cryogenically stored at 690000 Pa (100 psia) which is the minimum input pressure for the fuel cells and pumped out of the tank via a Helium gas feed system. Storage of the above volumes of reactants requires a LOX tank of radius .409 m, dry mass of 32 kg, and an LH_2 tank of radius .516 m, dry mass of 51 kg.

The PTLI stage of the Precursor Mission is identical to the PTLI stage of the Piloted Mission.

3.3.3.2 PTLI Thermal Control

The primary thermal control concerns on the propulsion stages are the cryogenic storage systems, the RL-10 engines, and the stage interior. The RL-10's are regeneratively cooled and have maximum rated burn times; therefore it is not necessary to provide an additional thermal control system for the engines. Thermal control of the stage interior is maintained passively through the applications of a reflective outer coating of silverized aluminum.

Insulation for the PTLI stage is designed to allow 0.175% fuel boiloff over a period of 40 days. Although the nominal duration of the PTLI stage's flight is only one tenth of this, it was decided to allow an extra month lest we miss the launch window for our second launch in a given manned flight.

The radius of the hydrogen tank outer surface are set at 2.8 m for a height of 6.0 m, with ellipsoidal endcaps of minor axis length 1.4 m. The radius and minor axis of the oxgen tanks are the same, although the cylindrical part of the LOX tanks is 0.6 m high.

Two hundred and forty-three layers of aluminized mylar are required to insulate the hydrogen tank, representing a total thickness of 17.44 cm, while the oxygen tank requires only 132 layers totalling 9.47 cm thickness. The total mass of the insulation is 1662.39 kilograms.

3.3.4 Guidance and Navigation System

For the EOR mission profile, the TLI stage will be launched separately, and must remain in orbit for at most 30 days before rendezvous and docking with the CM and ERM stages. In order to keep the TLI stage properly oriented for docking, a degree of attitude determination and control is necessary. Since the TLI stage is alone in LEO only, horizon sensors, sun sensors and gyroscopes are sufficient for attitude determination. Using two horizon sensor with a sun sensor and the IMU designed for the CM provides the required levels of redundancy for mission success. Processing of the attitude data will be done on the ground, and commands will be relayed to the PTLI stage to make attitude corrections.

The horizon sensors should be mounted with the scanning heads on opposite sides of the stage for both redundancy and precision. By subtracting the results of one CES from the other, bias and altitude errors can be eliminated. The output of the electronics of these sensors must then be sent to telemetry antennae to be downloaded to the earth. The sun sensor should be located on the outside of the PTLI stage, so that it is facing the sun.

By using the IMU designed for the CM and PLM, modularity is increased. Instead of powering all six gyroscopes, five of the six may be powered to meet the two levels of redundancy requirement for mission success. The IMU must be aligned with the spacecraft coordinate system, as outlined in Chapter 5 of Volume II.

Theses navigation aids ensure proper placement in the 6653km orbit. Once the chase vehicle is in the 6653km the docking sequences can begin.

Orbital Requirements

The orbital requirement is that the chase vehicle be placed on the parking orbit within the docking zone of the PTLI. The PTLI must launch to an altitude of 275km. This altitude ensures minimum ΔV's for rendezvous. It is also sufficiently above the atmosphere to avoid major orbit degradation.

Docking Configuration

The docking configuration includes a laser radar, video camera, visual target, and retro-reflectors. The radar system can determine range and range-rate fairly easily from any one of the passive reflectors. Altitude can be determined by differences in range to each reflector. Once the TV is close enough to the CV, if the field-of-view (FOV) is not great enough to encompass all three relfectors, angular orientation can be computed using just one retro-reflector.

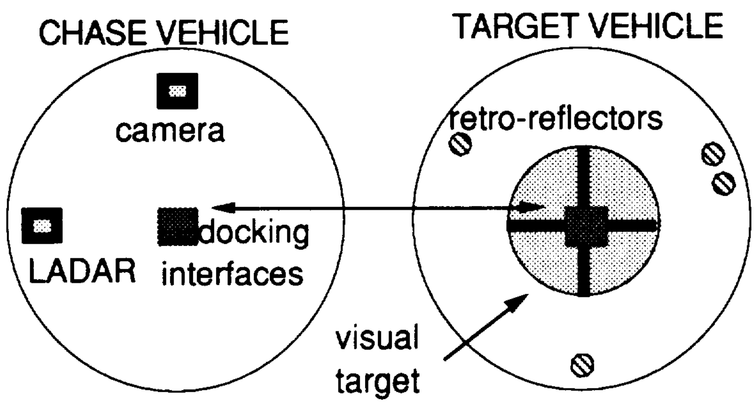


Figure 3-11
LADAR and Video Docking System

Figure 3-11 shows the physical configurration of the mating vehicles. The laser has a steerable beam allowing it a 40° FOV. This will enable the laser to "see" the retro-reflectors up to a distance of 6.5m. The precision with which LADAR is able to determine the agular orientation is given by equation 3-1.

$$AR = \arcsin\left(\frac{RR}{d}\right),\ AR = \frac{RR}{d} \text{ for small angles} \qquad (3\text{-}1)$$

where AR is the angular resolution, RR is range resolution, and d is the distance between retro-reflectors, 4.5m. So, for a respectable RR of ±5cm, AR will be 0.022° which is well under the 0.25° docking requirement.

The retro-reflector array is set in an equilateral-triangle with one of the vertices doubled up. In this configuration the LADAR can determine the proper orientation of the target vehicle such that the docking mechanisms interface correctly. Once the the chase vehicle is within the 6.5m range only one retro-reflectors is in view of the LADAR. At this time the vehicle is close enough that it can use INS. At the same time, the LADAR can track one of the retro-reflectors thus constantly calculating its relative attitude. Once the the vehicle is within several decimeters there is enough slop in the docking mechanism to mildly thrust the chase vehicle into the berthing interface.

Though it is useful to have range readings with a visual system, Brody explains that they are not imperative to successful pilot docking [Brody, 1987, Brody, 1990]. Adkins analyses different methods of augmenting the conventional display to increase the "intuitivenes" of docking the CV; in addition he covers some different visual markings systems which can yield accuracies better than those specified Vol. II, 5.3.3.1.1.

It has been decided to use autonomous docking in all rendezvous sequences since it will be a proven technology on the Precursor mission. Nevertheless, piloted docking could be performed very accurately, and thus will make a good back-up system for the automated docking.

3.3.5 Communications and Control System

The communications system on the PTLI stage consists of a low gain antenna system for use in tracking the PTLI and verifying its status from launch through docking operations. Details of the communications system on the PTLI stage are found in Volume II sections 4.2.1.

3.3.6 Status

The PTLI is the first major propulsion unit specifically designed by Project Columbiad. Because the majority of the stage is propulsion and tankage. Monitoring the status of the engines and the tanks, as well as the guidance system is of utmost importance. Once the injection burn is finished, the astronauts are on a minimum of a three day trip. A full system checkout must occur before the burn commences to ensure that the mission is viable.

3.3.6.1 Guidance

A typical guidance system on the kick stage which is still commanded by the crew capsule can include the following items:

- sun sensors(2)
- earth sensors(2)
- RCS
- telemetry and command radar
- small guidance computer

These items are constantly in need of checking for temperature and accuracy. There are self-checking routines built in and results can be compared to earth's to ensure accuracy and precision.

3.3.6.2 Propulsion

The monitoring of an engine and the accompanying tanks requires the placement of temperature, pressure, and power sensors at all critical spots. Especially important as well are accurate measurements of mass flow and fill percentage. The chamber temperature should be carefully measured because it can indicate a catastrophic failure about to happen. The RL-10 comes with attachments for monitors, and the tanks must be montiored for pressure, temperature, and fill level. In a propulsion system, the purpose of status is to predict and prevent catastrophic failures. Since propulsion is so vital to the sucess of the mission the monitoring must be complete, although not necessarily complex.

3.3.7 Subsystem Interfaces

Figure 3-12 graphically describes the interfaces between subsystems, showing power lines, data lines, and other pertinent information.

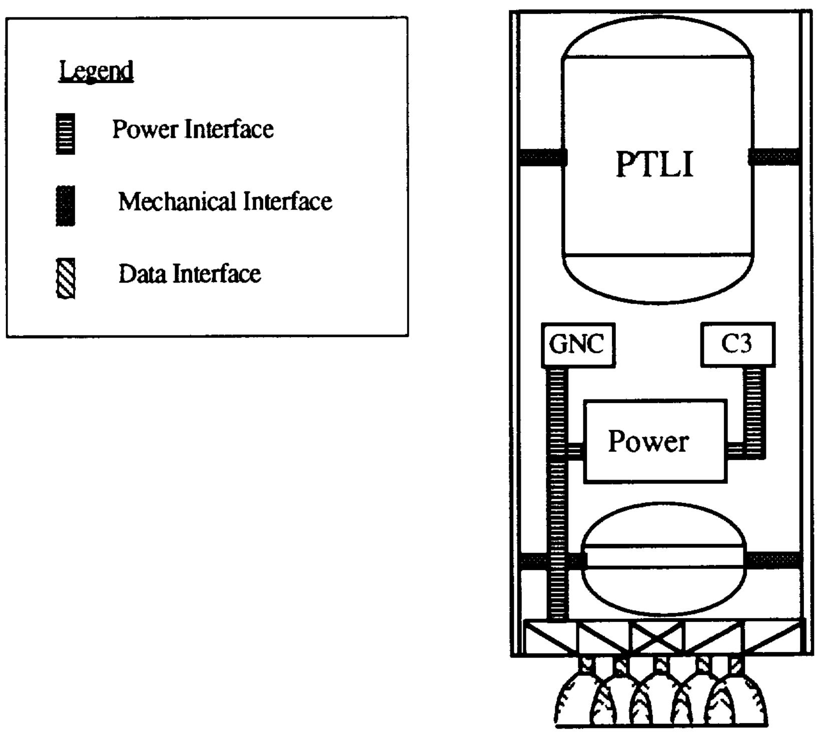


Figure 3-12
PTLI Subsystem Interfaces

3.3.7.1 Mechanical Interface

The mechanical interfaces consist of structural trusses that attach the tanks to outer shell.

3.3.7.2 Power Interface

The power interface supplies power from the fuel cells to propulsion subsystems, GNC subsystems, and C3 subsystems.

4. Lunar Braking Module

4.1 Stage Requirements and Operations

4.1.1 Requirements

The Lunar Braking Module (LBM) is the second stage of Project Columbiad following the Primary Trans-Lunar Injection Module (PTLI) as the first stage. The LBM is responsible for (1) final orbital insertion (2) upper orbital boost (3) completing trans-lunar injection after the PTLI has been exhausted (4) performing corrective burns during lunar transit (5) inserting the vehicle into lunar orbit and (6) providing most of the energy to brake from lunar orbit to lunar landing. Once these tasks have been completed, the LBM is jettisoned from the vehicle and crashes onto the surface of the Moon. Figure 4-1 gives a graphical summary of these tasks.

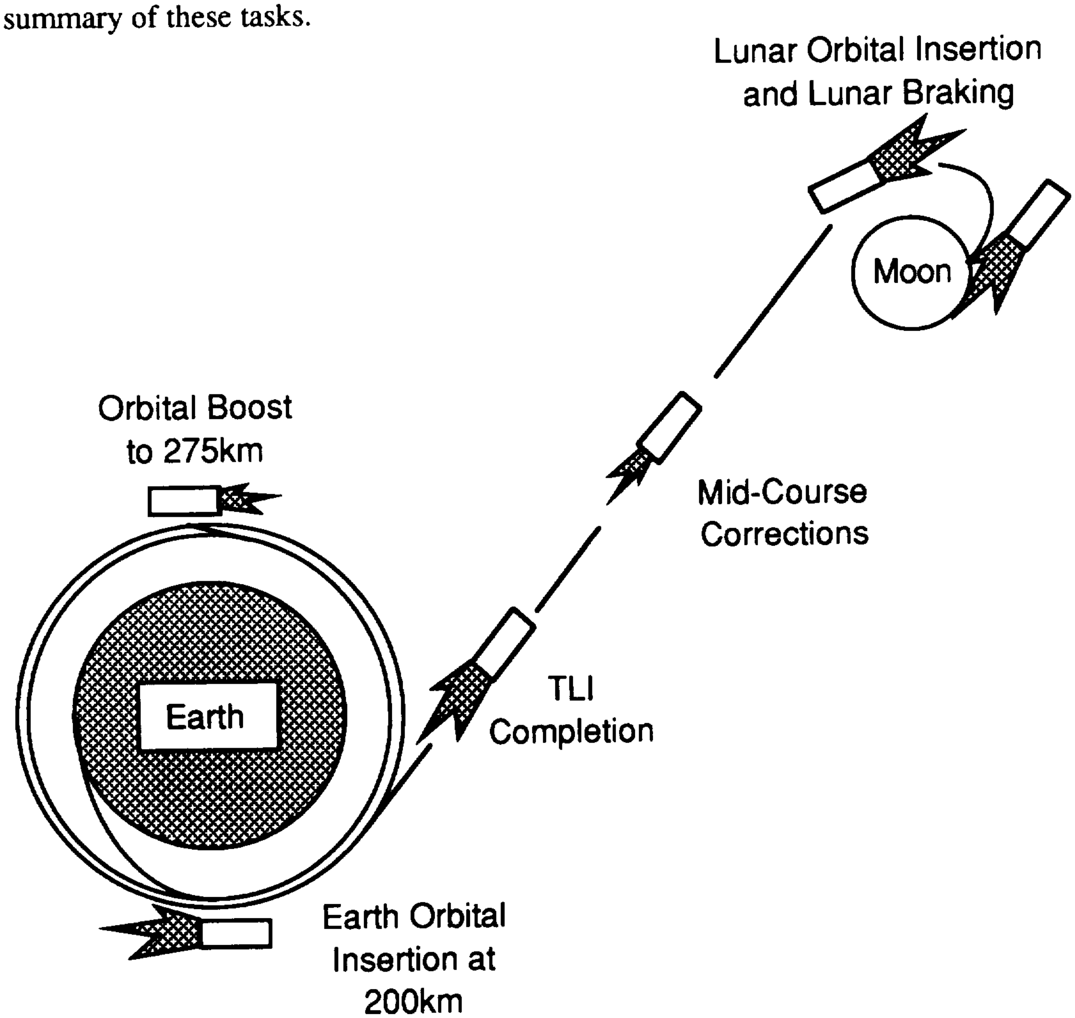


Figure 4-1
Summary of LBM Burns

The LBM is used in both the piloted and precursor missions and attaches either payload to its nose. The LBM performs the same mission for a payload of either the Earth Return Module (ERM) and Crew Module (CM) for the piloted mission, or the Payload Landing Module (PLM) for the precursor mission. The base of the LBM attaches to the launch vehicle during launch. Once the vehicle stages from the launch vehicle, the LBM completes low Earth orbital insertion. Next, the LBM rendezvous with the PTLI. Upon docking, the base of the LBM is mated to the nose of the PTLI. Once the PTLI has finished its mission requirements, the vehicle stages and the LBM assumes primary propulsive responsibilities for the mission.

The mission of the LBM is to propel the ERM/CM or PLM from near Earth to 600 m over the Moon's surface. The LBM consists of little more than propellants and engines inside a shell. The LBM has primary and secondary propulsion systems, a propulsion control system, a propellant feed system, and a status/monitoring system. Three RL10A-4 engines provide the primary means of propulsion for the entire stage. These engines burn liquid Oxygen and Hydrogen. The propulsion control system gimbals the main engines, opens and closes the valves when the engines start and stop, and adjusts the propellant mass rates for engine throttling. The propellant feed system consists of the lines and valves that lead from the propellant tanks to the engines. The status/monitoring system monitors the condition of the fuel and hardware. This status system consists solely of sensors. All LBM sensor data is fed to the Crew Module or Habitat computers for monitoring and interpretation. The power for the LBM is derived from another stage to which it is connected: the ERM in the piloted mission and the PLM in the precursor mission. Since the LBM is attached to other modules during its entire lifetime, it lacks inherent communications or guidance equipment; all of those functions are performed by either the ERM/CM package or the PLM/Habitat package.

4.1.2 Budgets

This section provides a budgetary breakdown of pertinent quantities for the Lunar Braking Module.

4.1.2.1 ΔV Budget

Table 4-1 shows the ΔV required and burn times for each phase of the LBM's operation. The RL10A-4 engines are throttlable, restartable, and gimbalable to attain the performance mandated by this thrusting profile. The engines are rated to burn for more than 4000 seconds, providing performance well beyond the scope of the LBM mission. A graphic display of these burns is shown earlier in Figure 4-1.

Table 4-1: RL10A-4 Engine Burns for LBM

Mission Phase	ΔV Required (m/s)	Burn Time (sec)
LEO Circularization at 200km	177	58
Upper Orbit Boost to 275km	44	14
Second Portion of Trans-Lunar Injection	680	181
Mid-Course Corrections in Transit	120	33 total
Entry into Low Lunar Orbit	1060	253
Lunar Braking(done with LLO entry)	1700	297
Total	**3781**	**836**

4.1.2.2 Mass Budget

Table 4-2 on the following page summarizes the major components of the LBM and some of their salient characteristics. The propellant masses have margins: 1) 1.5% of extra propellant is carried. 2) 0.2% of extra propellant is provided for anticipated cryogenic boiloff for four days using 18cm of Kapton insulation for hydrogen fuel tanks and 9cm for oxygen fuel tanks. While this thickness of insulation is more massive than the amount of fuel it stops from boiling off, the thick insulation allows more flexibility with scheduling and launch windows since it minimizes the propellant boiloff rate. The 0.2% factor was chosen so that the vehicle could stay in orbit for a month without compromising performance, and could stay much longer by delving into the extra propellant margin.

Table 4-2: Mass Breakdown for LBM

Component	Mass(kg)
Casing	2535
Propellant Tanks	1538
Tank Insulation	1800
Rocket Truss	267
LOX	47007
LH	8547
3 RL10A-4 Main Engines	504
6 Actuators	60
Helium Pressurization Tanks	20
Valves, piping, etc.	50
Staging Equipment, Wiring, etc.	120
LBM Total Dry Mass	**6894**
LBM Total Wet Mass	**62448**

4.1.2.3 Power Budget

The LBM has no inherent power of its own. It draws all of its power from the ERM or PLM. The stage has relatively small power requirements as summarized in Table 4-3.

Table 4-3: Power Allocation for LBM

Component	Power	
RL10A-4 Engine Startup/Valves/Shutdown	1500W	3*(25V, 20A)
Docking Radar and Cameras	250W	
Sensors	20W	
Power Required (continuous)	**20W**	
Power Required (peak)	**1670W**	

Table 4-4 breaks down volume considerations. The most important factor is the height of the module. If the total launching vehicle is too high, lateral forces and the structural response to these loads will cause additional unwanted problems. For a more detailed explanation of this criteria, see the launch vehicle Section 2.2 in Volume III. The propellant tanks include a 5% volume margin to minimize problems with not being able to fill the tanks completely.

Table 4-4: Volume Apportionment in LBM

Component	Volume (m^3)
3 RL10A-4 Main Engines (2.29m long* 1.2m diam.)	8.27
LOX, Tank, Insulation (3.28m long*5.4m diam. 2/1 end caps)	50
LH, Tank, Insulation (6.47m long*6.0m diam. 2/1 end caps)	149
Helium Pressurization Tanks	XXX
Valves, piping, etc.	0.5
Data and Power Lines	0.15
Sensors	0.25
LBM Total	**208**
LBM Maximum	**340**

4.1.3 Mission Profile

Figure 4-2 summarizes the time sequence of the LBM propellant burns. The LBM stages from the launch vehicle and nose cone 432 seconds after leaving the pad. Stagings occurs at 131km above the Earth's surface. At 967 seconds after launch, the vehicle reaches aposilon. The LBM burns 58 seconds to circularize its orbit at 200km. The LBM then boosts to a higher orbit of 275km. Rendezvous and docking maneuvers commence with the PTLI where the LBM's primary and the ERM/PLM's secondary propulsion systems are used sporadically. Once mating with the PTLI is finished, the PTLI begins trans-lunar injection. Then the vehicle stages and the LBM completes the injection. The trans-lunar flight requires three days, during which the primary and secondary propulsion systems fire occasionally to remain on course. When the vehicle arrives near the Moon, the LBM burns a final time for 550 seconds to slow enough not only to come into lunar orbit, but also brake lunar orbital velocity.

The LBM stages from the rest of the vehicle 600m above the lunar surface and falls to the lunar surface. The ERM or PLM completes the landing. The safety of the lunar base is insured by staging the LBM about one kilometer downrange of where the base is.

A more thorough description of the mission trajectory and stage burns can be found in Section 5.3 of Volume I. Figure 4-2 is a burn time line for the LBM showing the sequence, spacing, and duration of each burn.

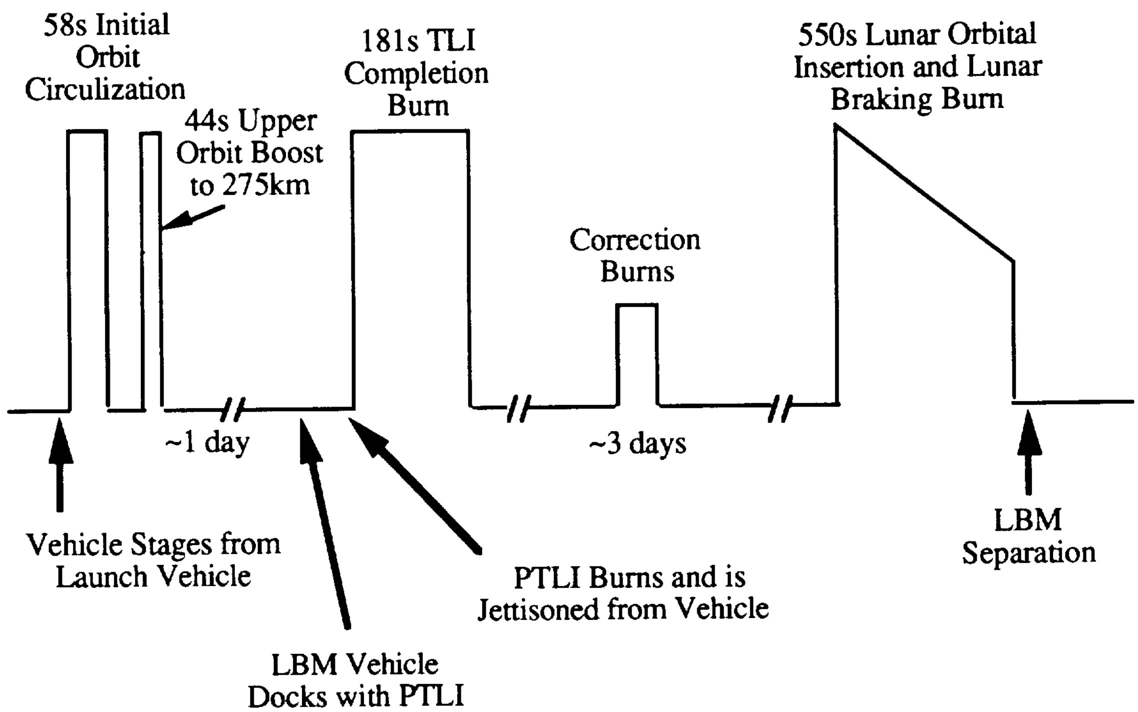


Figure 4-2
LBM Burn Timeline

4.1.4 Abort Options

The LBM supports abort options for both the precursor and piloted missions. For the precursor mission, successful aborts result in the delivery of the payload to the lunar surface. For the piloted mission, a successful abort results in the completion of the planned mission and/or ensures crew survival.

4.1.4.1 Trans-Lunar Injection Abort

The Lunar Braking Module possesses a single engine-out capability (2 out of 3 engines operable) for its portion of the trans-lunar injection burn.

4.1.4.2 Trans-Lunar Abort (Piloted Mission)

During Trans-Lunar coast, several abort modes are available depending upon the timing, nature, and severity of the emergency. A direct return abort can be initiated at any time during the outbound leg. The primary propulsion systems of the LBM supports this abort mode. The LBM fires to cancel the forward velocity of the spacecraft and place the vehicle on a return trajectory. A second abort mode (Near Lunar Abort) delays the initiation of an abort propulsive burn until the spacecraft is within the vicinity of the Moon (3 days out from Earth). Near the Moon, while behind the visible face, the primary propulsion system of the LBM burns to place the spacecraft onto an earth-return trajectory. This abort mode places less demanding requirements upon the spacecraft propulsion and guidance systems, and would be used if the extra transit time needed to complete such an abort were deemed available.

4.1.4.3 Lunar Orbit Insertion Abort

The Lunar Braking Module possesses a single engine-out capability (2 out of 3 engines operable) during the Lunar Orbit Insertion (LOI) burn. In the event of a decision to abort landing operations at this point, the ERM (along with the remaining propulsive capability of the LBM, if needed) injects the spacecraft into an Earth-return trajectory using the Near Lunar Abort mode (piloted mission only).

4.1.4.4 Descent Abort

The LBM is capable of completing its descent propulsion burn with a single engine-out failure, although the fuel reserve available (in the ERM/PLM) for final hover is minimized, decreasing the time available for last minute flightpath corrections.

4.2 Stage Design

4.2.1 Configuration

The Lunar Braking Module is cylindrically shaped, 6m in diameter and 13.25m long. The LBM contains a liquid Hydrogen tank, a liquid Oxygen tank, three RL10A-4 engines, and structure. Both propellant tanks are in the shape of a cylinder with two to one end caps. The end caps are four times as wide as they are high. The LOX tank is 3.28m long and 2.7m in diameter including 0.2m of Kapton insulation. The LH2 tank is 6.47m long and 3.0m in diameter also including 0.2m of insulation. The three RL10A-4 main engines are in a cluster. The outer 0.2m of the tank surfaces is Kapton cryogenic insulation. The RL10 engines are mounted 4° off the PLM's cylindrical axis. This off-centering choice is made to align the thrust closer to the center of mass. The line from the center of mass to the RL10A-4 engines is near 5° off the cylindrical axis. The 4° parameter allows the engines to thrust along the cylindrical axis at maximum gimbaling. If an engine is out, there is a -3° to +5° gimbaling capability around the center of mass. Thrusting through the center of mass puts the thrust 5° off the cylindrical axis. This 5° parameter lowers the thrust of the PLM by cos(5°) to 99.6% of ideal in the case of an engine out. Helium tanks pressurize the Hydrogen and Oxygen tanks. The RL10A-4 engines from the PLM/ERM fit into an interstitial space above the Oxygen tank. The modules fit together in this fashion to incorporate the nozzle protection skirt into the LBM. Grooves in the skin of the LBM accommodate the landing struts from the ERM/PLM. Grooves are also cut at the top of the LBM for the RCS exhaust from the PLM and ERM. Since the LBM has no RCS units, the ERM/PLM must. To generate a thrust rearward, the ERM/PLM must thrust into the LBM. To avoid problems burning holes in the hydrogen tank, ducts are provide to divert the exhaust around the LBM. This configuration is shown in Figure 4-3 on the next page.

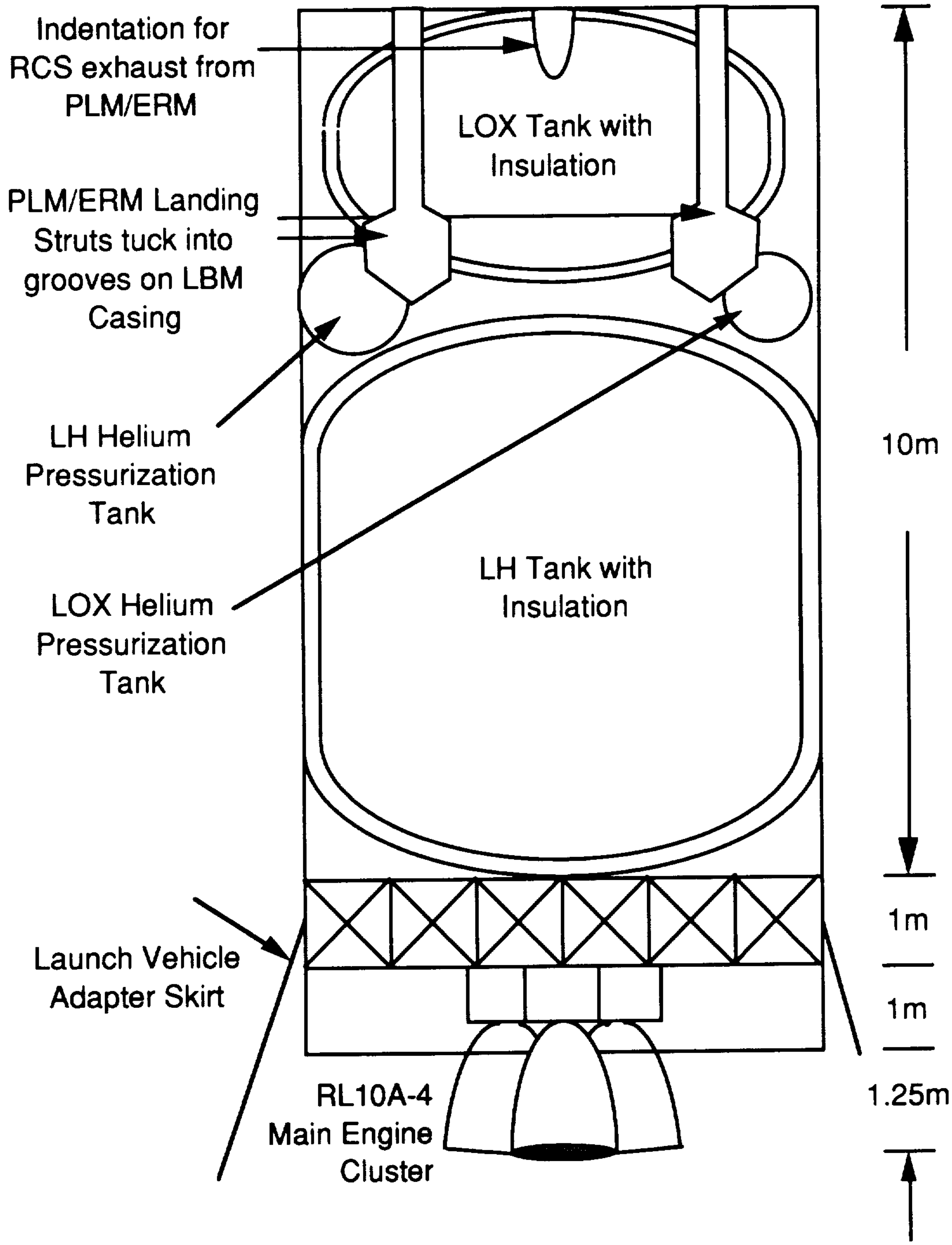


Figure 4-3
Internal Configuration of LBM

4.2.2 Vehicle Interfaces

This section documents the interfaces for the LBM, including mechanical and data interfaces, and power interface where appropriate.

4.2.2.1 Launch Vehicle

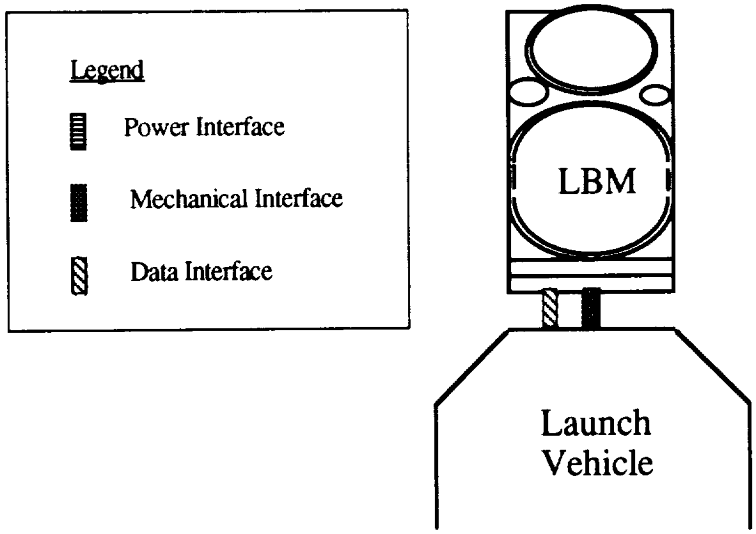


Figure 4-4
LBM/Launch Vehicle Interface

4.2.2.1.1 Mechanical Interface
The mechanical interface between the LBM and the launch vehicle consists of an explosive skirt which is attached to the top of the truss located at the lower end of the LBM. The skirt spacer adapts the 6m LBM diameter to the 8m launch vehicle diameter.

4.2.2.1.2 Data Interface
The data interface between the LBM and the launch vehicle is to monitor launch vehicle status.

4.2.2.2 Primary Trans-Lunar Injection Stage

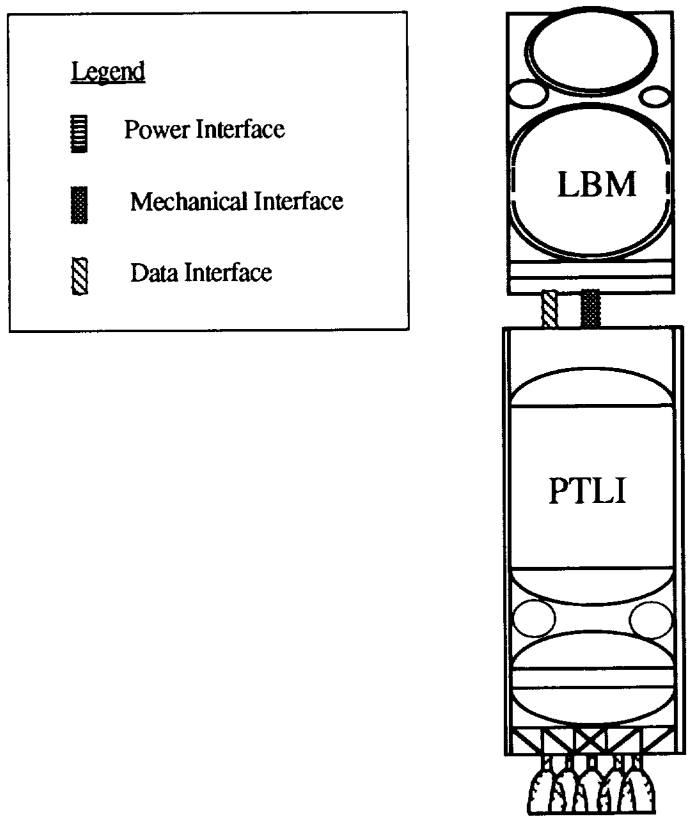


Figure 4-5
LBM/PTLI Interface

4.2.2.2.1 Mechanical Interface

The mechanical interface between the LBM and PTLI consists of docking latches for rendezvous and docking. For stage separation, the interface has explosive bolts.

4.2.2.2.2 Data Interface

The data interface between the LBM and PTLI is to transmit information to the CM which monitors the status of the tanks and engines.

4.2.2.3 Earth Return Module

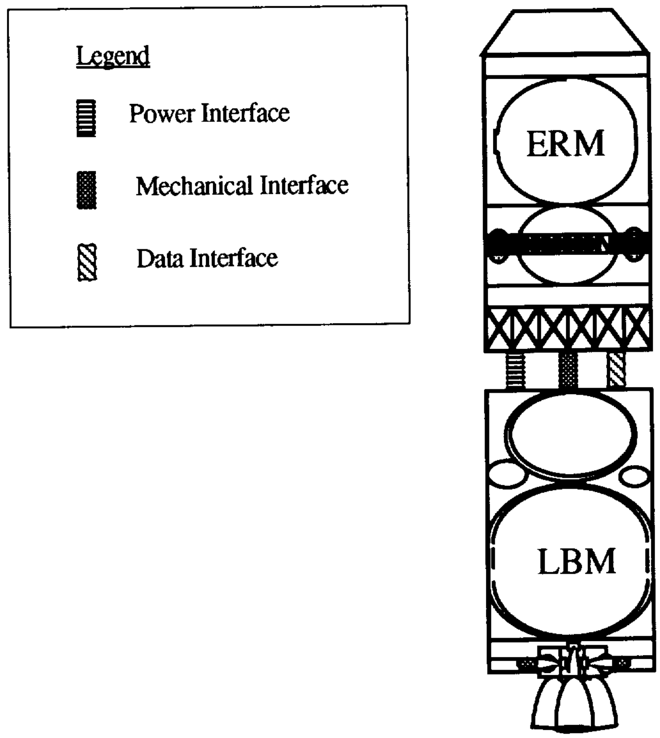


Figure 4-6
LBM/ERM Interface

4.2.2.3.1 Mechanical Interface
The mechanical interface between the LBM and ERM consists of explosive bolts for stage separation.

4.2.2.3.2 Power Interface
The power interface between the LBM and ERM connects the fuel cells in the ERM to all subsystems in LBM.

4.2.2.3.3 Data Interface
The data interface is a database between the LBM and ERM which transmits LBM status to the computers in the CM.

4.2.2.4 Payload Landing Module

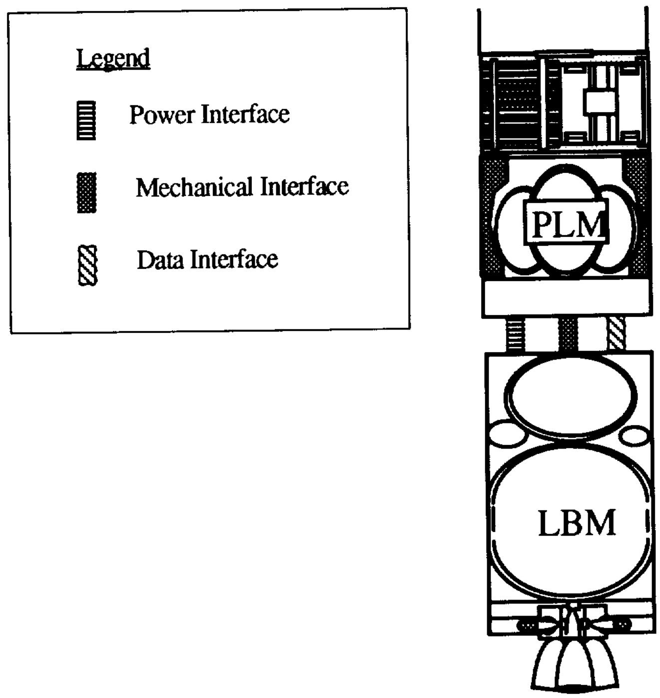


Figure 4-7
LBM/PLM Interface

4.2.2.4.1 Mechanical Interface

The mechanical interface between the LBM and PLM consists of explosive bolts for stage separation.

4.2.2.4.2 Power Interface

The power interface between the LBM and PLM connects the fuel cells in the PLM to all subsystems in LBM.

4.2.2.4.3 Data Interface

The data interface is a database between the LBM and PLM which transmits LBM status to the computers in the Habitat. The data link also provides a connection between the LBM and PLM for command and engine control.

4.3 Subsystem Design

4.3.1. LBM Structural Design

A summary of the LBM structural design is included here as a service to the reader. For those interested in the methods of the design or desire to understand the structural trade-off that were involved in the choice of this structural configuration you are referred to section 2.1.4 and 2.2.2 in Volume II.

The geometrical description and masses are summarized below in Tabees 4-4, 4-5, and 4-6.

Table 4-4: LBM Hydrogen Tank Design Parameters

Hydrogen Tank	
Hydrogen Mass	8553.85
Hydrogen Volume	120.48
Hydrogen Tank Volume	126.50
Hydrogen Tank Radius	2.80
Hydrogen Tank Cap Radius	1.40
Hydrogen Tank Cap Volume	45.98
Hydrogen Tank Main Volume	80.52
Hydrogen Tank Main Height	3.27
Hydrogen Tank Cap Eccentricity	0.87
Hydrogen Tank Cap Area	57.39
Hydrogen Tank Body Area	57.52
Hydrogen Tank Area	114.91
Hydrogen Tank Wall Thickness	0.0010
Hydrogen Tank Structure Mass	178.29
Hydrogen Tank Coating Thickness	0.0010
Hydrogen Tank Coating Mass	873.32
Hydrogen Tank Height	6.07
Hydrogen Tank Insulation Mass	1241.04
Hydrogen Tank Mass	1051.61

Table 4-5: LBM Oxygen Tank Design Parameters

Oxygen Tank	
Oxygen Mass	47046.15
Oxygen Volume	38.25
Oxygen Tank Volume	40.16
Oxygen Tank Radius	2.50
Oxygen Tank Cap Radius	1.25
Oxygen Tank Cap Volume	32.72
Oxygen Tank Main Volume	7.44
Oxygen Tank Main Height	0.38
Oxygen Tank Cap Eccentricity	0.87
Oxygen Tank Cap Area	45.75
Oxygen Tank Body Area	5.95
Oxygen Tank Area	51.70
Oxygen Tank Wall Thickness	0.0012
Oxygen Tank Structure Mass	93.13
Oxygen Tank Coating Thickness	0.0010
Oxygen Tank Coating Mass	392.94
Oxygen Tank Height	2.88
Oxygen Tank Insulation Mass	558.39
Oxygen Tank Mass	486.08

Table 4-6: LBM Configuration Summary

Configuration	
Stage Radius	3
Total Height	13.20
Casing Height	10.95
Insulation Mass	1799
Casing Mass	2345
Rocket Truss Mass	267
Tank Mass	1538
Structural Mass	4150
Engine Mass	501
Stage Dry Mass	6450
Stage Wet Mass	62050
Vehicle Wet Mass	90871
Structural Mass Fraction	7%
Structural Fuel Fraction	7.5%

4.3.2 Propulsion

4.3.2.1 Primary Propulsion System

The primary propulsion system of LBM stage is shown in Figure 4-8. It consists of three RL10A-4 engines rated at 92,518 N nominal thrust and operating each at a 5.5:1 mixture ratio of oxidizer to fuel. The net positive suction head (NPSH) required by the engine turbopumps is provided by pressurizing the vehicle propellant tanks with helium gas at 272 atm. Propellants are delivered to the main engine turbopumps through feed ducts from the vehicle propellant tanks. The feed ducts contain flex joints to accomodate engine gimbaling and are overwrapped with a three-layer, double aluminized Kapton radiation shield.

The primary propulsion engines run on a bipropellant combination of liquid oxygen oxidizer and liquid hydrogen fuel. Both propellant tanks are cylindrical with semi-spherical endcaps, and are constructed of a thin steel core overwrapped with pre-stressed graphite composite fibers and a 20 cm layer of aluminized Kapton insulation. The oxidizer tank is 2.88 m tall and 5 m in diameter; the fuel tank is 6.07 m tall and 5.6 m in diameter.

Pneumatically actuated prevalves located at the propellant tank outlets provide series redundant backup for the engine inlet shutof valves. A parallel set of pyro valves and solenoid valves upstream of the pneumatic actuation control solenoid valves provides two-failure tolerance against inadvertent opening of the engine inlet shutoff valves. The pyro valves will be fired open after the LBM stage is deployed a safe distance from the PTLI stage. The system also has manual fill and drain valves to load propellant and pressurant gas into the system, as well as additional manual valves for system leak checking on both sides of the pyro-isolation valves and regulators. Check valves insure that the fuel and oxidizer can never mix anywhere in the system, except in the engine. Finally, pressure transducers, filters, temperature sensors, and line and component heaters are provided to ensure proper subsystem operation. A mass distribution of the entire primary propulsion system is given in Table 4-7.

Table 4-7: Mass Distribution of LBM Primary Propulsion System

COMPONENT	MASS [kg]
Empty Fuel Tank	1,051
Fuel Mass	8,518
Empty Oxidizer Tanks	486
Oxidizer Mass	47,036
Empty Helium Tanks	109
Helium Mass	102
Monitoring equipment	20 (estimated)
Propellant lines	26 (estimated)
Valves	39
Engine mass (3 RL10A-4 engines)	504
TOTAL FUELED WEIGHT	57,891 kg

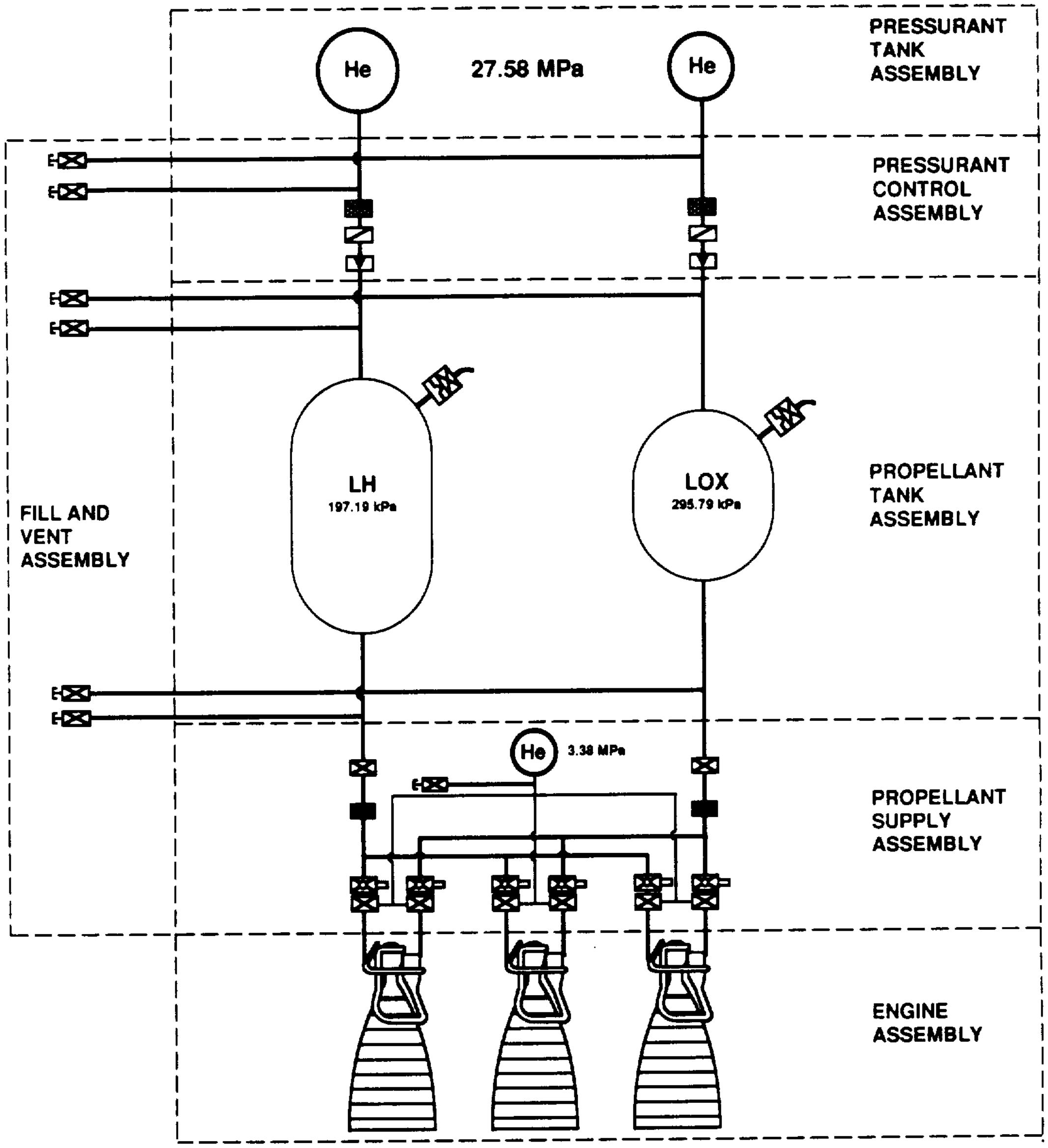


Figure 4-8
LBM Primary Propulsion System

4.3.2.2 Reaction Control System

The reaction control system of the LBM stage consists of two redundant subsystems configured as shown in Figure 4-9. Each subsystem consists of 8 R-4D thrusters operating on a 1.65:1 mixture ratio of oxidizer to fuel and fed by two propellant tanks. The thrusters are divided into quadruple clusters which are placed along the periphery of the spacecraft, making a total of 16 thrusters and four propellant tanks for the complete system.

The system utilizes a bipropellant combination of nitrogen tetroxide oxidizer and monometylhydrazine fuel. The propellants are stored in separate spherical tanks of identical size; each tank is 0.76 m in diameter. Both tanks are constructed of a thin steel core overwrapped with pre-stressed graphite composite fibers; no thermal insulation material is required. Propellants are equipped with a Teflon diaphragm positive expulsion device which insures efficient tank evacuation.

A pressurant tank stores helium at about 272 atm, and a quad redundant regulator — coupled with a burst disk and relief valve— regulates flow. Together, they insure a 15 atm feed pressure to the propellant tanks, even after any single regulator failure. There are burst disks and pyrotechnically actuated squib valves to isolate propellants from the engine (and high pressure gas from the propellant tanks) until the system is ready for operation. This system also has manual fill and drain valves to load propellant and pressurant gas into the system, as well as additional manual valves for system leak checking on both sides of the pyro-isolation valves and regulators. Check valves insure that the fuel and oxidizer can never mix anywhere in the system, except in the engine. Finally, pressure transducers, filters, temperature sensors, and line and component heaters are provided to ensure proper subsystem operation. A mass disrtibution of reaction control system components is given in Table 4-8.

Table 4-8: Mass Distribution of LBM Secondary Propulsion System

COMPONENT	MASS [kg]
Empty Fuel Tanks	20
Fuel Mass	192
Empty Oxidizer Tanks	20
Oxidizer Mass	317
Empty Helium Tanks	6
Helium Mass	2
Monitoring equipment	20 (estimated)
Propellant lines	26 (estimated)
Valves	62
Engine mass (16 R4-D engines)	60
TOTAL FUELED WEIGHT	725 kg

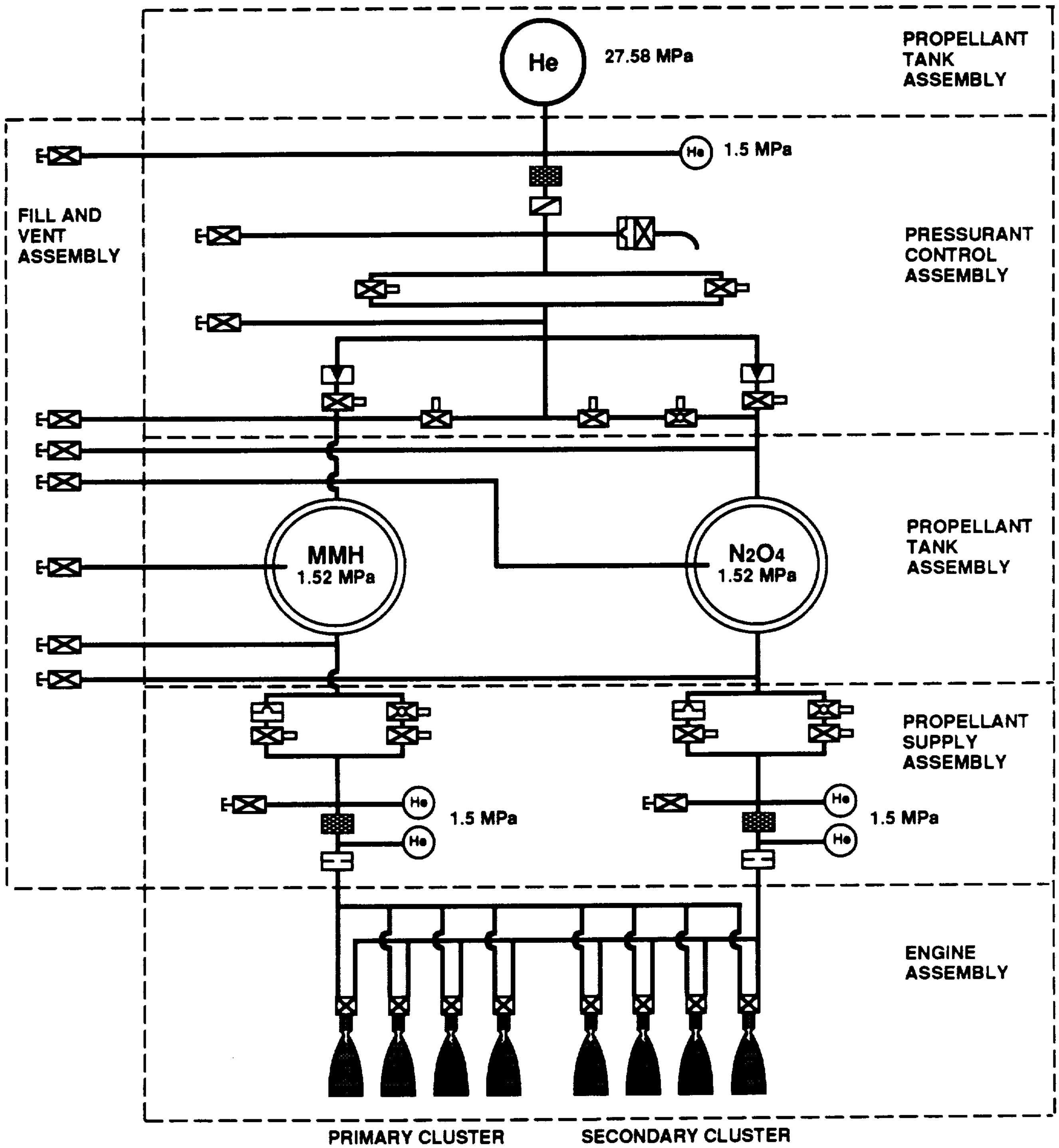


Figure 4-9

LBM Secondary Propulsion System

4.3.3 Power and Thermal Control

4.3.3.1 LBM Power Supply

The LBM has many of the same propulsion stage needs as the PTLI. It is estimated to require 1 kW of power over the 4 day duration of its use. The LBM of the Precursor Mission will draw its power from the fuel cell system of SLURPP on board the PLM stage immediately above the LBM. The LBM of the Piloted Mission draws its power from the ERM stage immediately above the LBM. The LBM's of the two missions are identical. The LBM power can be supplied by adding 37.7 kg of reactant to the PLM propellant tanks and by using 6.8 kg of the SLURPP fuel cells, or by adding 37.7 kg of reactant to the ERM fuel cell reactant tanks and by using 6.8 kg of the ERM fuel cells. The additional reactant breaks down as 33.5 kg O_2 and 4.18 kg H_2, or as .029 m^3 of O_2 and .059 m^3 of H_2.

4.3.3.2 LBM Thermal Control

The primary thermal control concerns on the propulsion stages are the cryogenic storage systems, the RL-10 engines, and the stage interior. The RL-10's are regeneratively cooled and have maximum rated burn times; therefore it is not necessary to provide an additional thermal control system for the engines. Thermal control of the stage interior is maintained passively through the applications of a reflective outer coating of silverized aluminum. The cryogenic systems are thoroughly described in section 6.3.

Insulation for the LBM stage is designed to allow 0.175% fuel mass boiloff over a period of 4 days.

The radius of the hydrogen tank outer surface are set at 2.8 m for a height of 3.27 m, with ellipsoidal endcaps of minor axis length 1.4 m. The radius and minor axis of the oxygen tanks are the same, although the cylindrical part of the LOX tanks is 0.38 m high.

One hundred and ninety-four layers of aluminized mylar are required to insulate the hydrogen tank, representing a total thickness of 13.92 cm, while the oxygen tank requires only 124 layers totalling 8.90 cm thickness. The total mass of the insulation is 958.97 kilograms.

4.3.4 Communications and Control System

Information from sensors required to be relayed to status monitoring systems will interface with the databus of the ERM and CM stages which will, in turn, relay the necessary information to Earth and onboard status monitoring systems.

4.3.5 Status

The LBM has many of the same requirements as the PTLI. Before beginning the final braking burn, it is important to have a complete update of the status for every system onboard and to have examined the systems on the moon. Before the burn, this is the last easy abort possiblity. Now abort is a tricky issue and any major propulsive failure makes it impossible.

4.3.6 Subsystem Interfaces

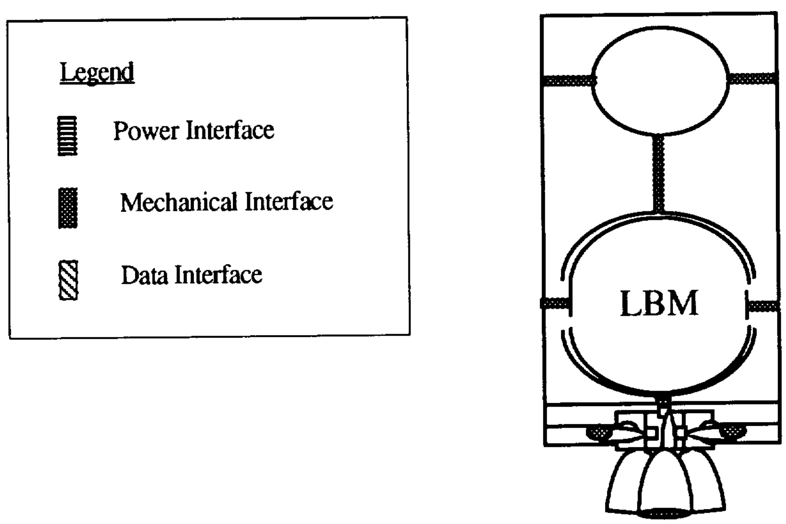


Figure 4-10
LBM Subsystem Interfaces

The mechanical interfaces in the LBM serve two purposes. There are structural truss that attach the tanks to the outer shell and also propulsion lines that feed propellant to the engines.

5. Earth Return Module

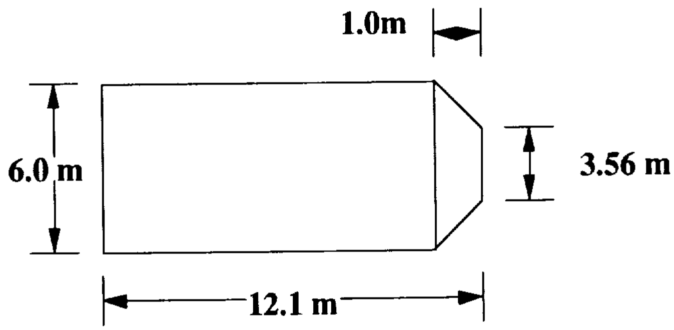
5.1 Stage Requirements and Operations

5.1.1 Requirements

The Earth Return Module (ERM) must adhere to a set of predetermined requirements in order to successfully perform its mission. These requirements fall into three different categories: transportation requirements, configuration requirements, and support requirements.

The transportation requirements deal with what maneuvers the propulsion system of the ERM must perform. The primary requirement in this category is that the ERM must provide a means of transporting the Crew Module (CM) from the Moon to the Earth. The ERM must also provide a means of hovering and landing the CM onto the lunar surface, and must be able propel the CM from the surface of the Moon to lunar orbit and a subsequent trans-Earth injection trajectory. These maneuvers will be presented in greater detail in the section regarding the Mission Profile.

The configuration requirements deal with the physical volume and mass requirements that the ERM must satisfy. The current design of the launch system imposes a maximum of six meters on the diameter of the ERM. In addition, there has been alloted a maximum height of 12.1 meters for the ERM in their configuration of the fourth launch. The current height of the ERM design calls for a height of 9.97 meters. Because there will be no fairing between the ERM and the CM, an extra requirement has been imposed upon the configuration of that interface so that the aerodynamic loads imposed upon the ERM-CM interface during launch will be reduced: the last meter of the ERM must taper from 6 meters to 3.56 meters, the diameter of the CM. Functionally, the ERM must be modular to a certain degree, promoting ease of maintenance and reusability, and it must possess the ability to land on the Moon (i.e. landing gear). The mass requirement of the ERM is presently 26,210 kg, the number calculated by Systems. However, this is more of a goal than a requirement, as the top-level requirement is that the entire fourth launch must weigh under 91 metric tonnes. The configuration requirements of the ERM are summarized visually in Figure 5-1.



Weight Goal: 26,210 kg

Figure 5-1
ERM Configuration Requirements

The support requirements deal with what components aboard the ERM must be designed to support other aspects of the mission. First of all, the power system aboard the ERM must be able to handle most of the power requirements of the CM and the LBM in addition to the power requirements of the ERM itself. Second, the ERM must contain enough storage space so as to accommodate extra supplies and components for the manned mission (payload). Finally, the ERM must be able to have enough mass and volume capacity to contain components that would otherwise be on other modules (like the CM) but should be placed on the ERM because of volume constraints or better location.

5.1.2 Budgets

There are five resources that must be budgeted properly if the ERM is to complete its mission without the need for refueling. These are the delta-vee (velocity change) budget, the propellant budget, the power budget, the mass budget, and the volume budget.

The ΔV budget consists of the different velocity changes that must be executed by the ERM during its mission. The details of these maneuvers are contained within the Mission Profile section, and the numbers are contained in Table 5-1.

The propellant budget consists of how much propellant of each type is needed to perform the maneuvers stated in the velocity change budget. This budget is extremely important because it has a large bearing on the weight of the ERM due to the large amount of propellant involved. The propellant for the main engines consists of liquid hydrogen (LH2) as a fuel and liquid oxygen (LOX) as an oxidizer, and the propellant for the Reaction Control System consists of MMH as a fuel and N2O4 as an oxidizer. Note that the numbers given in Table 5-1 consist of the entire propellant weight required, that the propellant masses include an extra 2% to account for boil-off, and that the Lift-off to LLO and LLO to TEI burn numbers have been combined into figures for a single burn.

Table 5-1: ΔV and Propellant Budgets

Maneuver	Burn By	ΔV (m/s)	Duration (s)	Propel. (kg)
Lunar Land.	Main Engines	500	55	3551.24
Lift-off to LLO	Main Engines	2200	212	13627.58
LLO to TEI	Main Engines	1060	incl. above	incl. above
Midcourse Adj.	Main Engines	240	short pulses	956

The power budget details the power alloted to each component of the ERM, including the Crew Module and the LBM. The power budget introduces an additional complication in that the power requirements of the system vary with respect to time. As a result of this, in addition to tabulating the power needs of each component, seen in Table 5-2, a power-time curve has been provided (Figure 5-2). Note that the totals of the power budget already contain margins of safety (2-5%).

Table 5-2: Power Budget

Component	Subsystem	Number	Power (W)
Star Trackers	GNC	3	5 each
Sun Sensor	GNC	1	2
Radar Altimeters	GNC	3	100 each
Antenna Beacons	GNC	2	20 each
Main Engine Burn	Propulsion	3	35 each
RCS Burn	Propulsion	16	10 each
Crew Module	Crew Capsules	1	4767 Max.
LBM	Prop. Stages	1	478 Max.
Explosive Bolts	Prop. Stages	4	100
Landing Gear	Structures	3	150
Monitoring	Status	1	10
TOTAL (Max):			**6127 Watts**

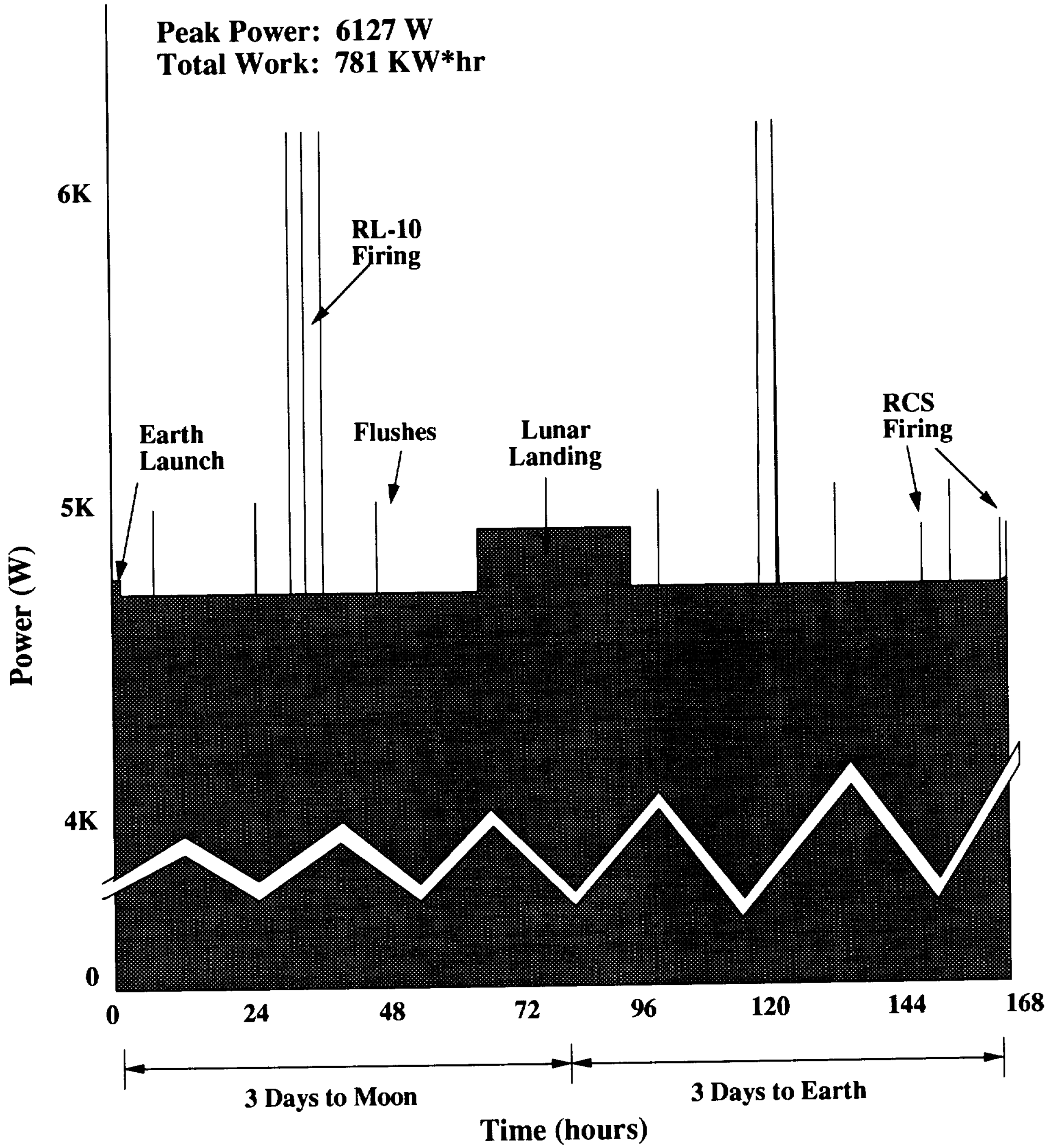


Figure 5-2
Power-Time Curve

An important thing to notice about the power-time curve in Figure 5-2 is that it consists of a series of spikes that characterize engine firings, etc., combined with the constant loads imposed by the CM. Therefore, the power system of the ERM must be able to provide a steady power flow and possess enough reserve power to handle the occasional spikes.

The mass and volume budgets determine the actual mass and configuration of the vehicle, and they also reveal whether or not the present design has met the specifications detailed in the Requirements section.

The primary physical properties of the ERM are as follows:

- Diameter = 6m
- Current Length = 9.97m
- Current Volume = 280.865m3
- Mass of Structure and Payload (calculated) = 8285.74 kg
- Mass of Propellant (calculated) = 18134.8 kg
- Total Mass (calculated) = 26420.54 kg

The mass and volume of each individual component of the ERM are contained within Table 5-3. Note that the mass and volume figures are totals for all components present, complete with margins.

Table 5-3: Mass and Volume Budget

Component	Subsystem	Number	Mass (kg)	Volume (m3)
Star Trackers	GNC	4	100.0	0.400
Sun Sensor	GNC	1	1.0	0.001
Radar Alt.	GNC	3	90.0	0.300
Antenna Beac.	GNC	2	6.0	0.020
Main Engines	Propulsion	3	504.0	63.62
RCS	Propulsion	16	1139.74	1.46
LH2	Prop./STP	1	2576.8	37.4
LOX	Prop./STP	1	14602	13.214
Tanks & Misc	Propulsion	2	678	95.52
Insulation	Propulsion	1	823	see 'Tanks'
Power System	PTC	1	467	13.416
Outer Structure	STP	1	1666	thin skin
Internal Struct.	STP	1	267	28.274
Landing Gear	STP	3	500	outside
Payload	Surface Pay.	1	3000	27.245
TOTAL:			**26420.54**	**280.865**

Note: Figures for 'Tanks & Misc.' include pipelines, valves, pressurization system, and main tanks. Figures for LH2 and LOX include propellant only. Figures for 'RCS' and 'Power System' contain amounts for the entire subsystem.

5.1.3 Mission Profile

The Mission Profile of the ERM consists of four operations: initial separation from the Lunar Braking Module (LBM), hovering and landing on the lunar surface, launch from the lunar surface, and subsequent trans-Earth injection. Completion of these operations are what dictates the requirements on the subsystems of the ERM.

5.1.3.1 Separation from LBM

Prior to separation from the LBM, the ERM will execute 3 to 5 RCS burns for midcourse corrections. The separation of the ERM from the LBM will take place while the vehicle is descending from LLO to the lunar surface. At an altitude of about 2800 m, a signal from the CM will activate the explosive bolts situated in the LBM-ERM interface, and the connection will be severed. The LBM will execute a main engine burn (see LBM section for details) to bring it away from the ERM, where it will land on the moon at a calculated 1,697 m away from where it was staged. The ERM will then begin the hovering maneuver that will land it onto the lunar surface. The operation is visually detailed in Figure 5-3.

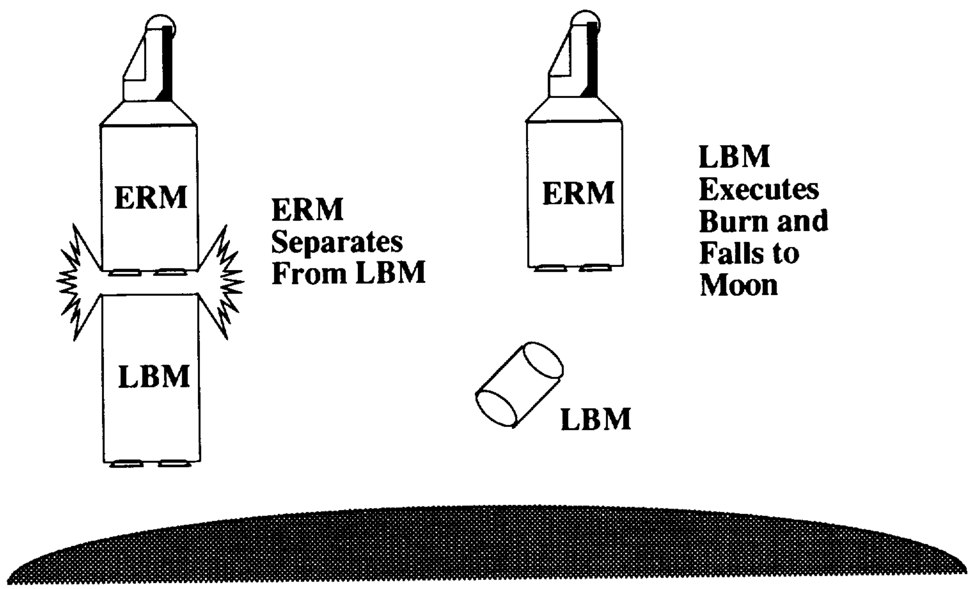


Figure 5-3
Separation From LBM

5.1.3.2 Hover/Landing

After the ERM-LBM separation, the ERM will begin its hovering maneuver. The ERM descends the final distance to the lunar surface on a slanted trajectory. This path insures that the landing site and lunar habitat are not jeopardized by the LBM falling to the surface. The radar altimeter will be activated and the main engines will execute a burn of a 55 second duration that will comprise a ΔV of about 500 m/s. This should be sufficient to land the ERM onto the lunar surface. At about 25m above the lunar surface the landing gear will be deployed. Position adjustment will be accomplished by gimbaling the main engines, and the desired landing position of the ERM will be determined through the use of the antenna beacons aboard the ERM. This operation is represented in Figure 5-4 on the following page. After the ERM has landed, it will proceed to power down and shut off all non-essential systems.

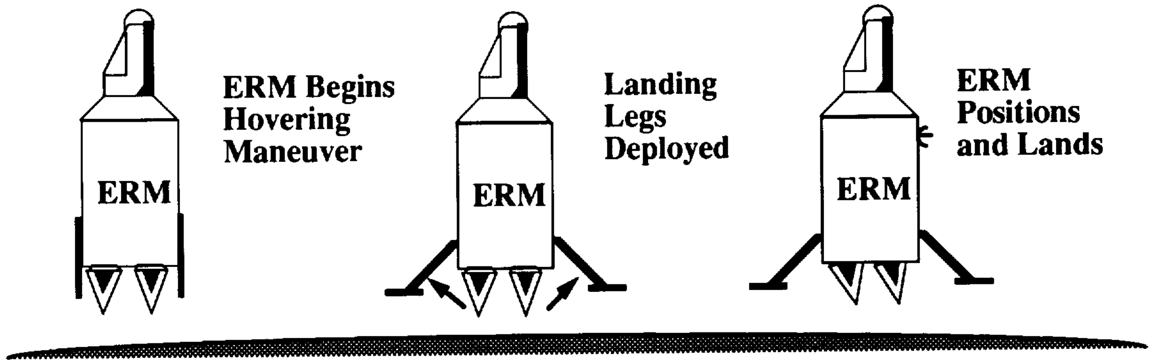


Figure 5-4
ERM Hover/Landing

5.1.3.3 Launch from Lunar Surface

Before the launch from the lunar surface occurs, the ERM will power up to full capacity (see Figure 5-2 for exact details). After all systems have checked out, a main engine burn with a ΔV of 2200 m/s will be executed and the landing gear will be jettisoned. This will launch the ERM from the surface and put it into LLO again. When LLO is reached, the ERM will wait for the nearest launch window to the Earth and then execute a main engine burn with a ΔV of 1060 m/s. The combined duration of these two burns is 212 seconds. This operation is represented in Figure 5-5.

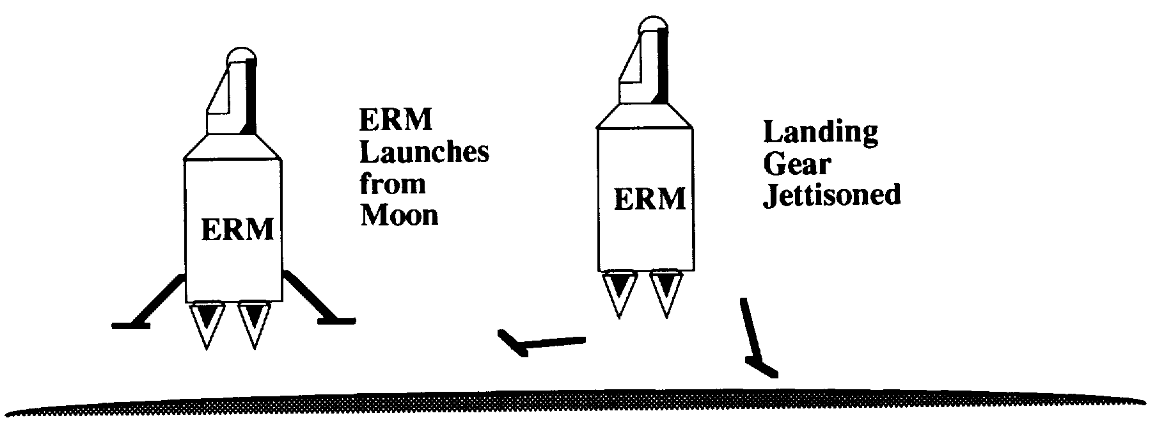


Figure 5-5
Launch From Lunar Surface

5.1.3.4 Trans-Earth Injection

The trans-Earth injection maneuver is accomplished by the burn stated in the section above. During the transit time to the Earth, positioning will be accomplished through the use of three star trackers and a sun sensor, and adjustments will be made with three RCS burns made at appropriate times along the trip. Just before (about 20 minutes) the vehicle reaches Earth's atmosphere, the ERM will be jettisoned from the CM through the use of explosive bolts, and the ERM will execute an RCS burn to bring it away from the CM, and will burn up in the Earth's atmosphere. This operation is represented in Figure 5-6.

Figure 5-6
Trans-Earth Injection

Each of these operations is performed without the need for crew interaction, with the exception of the lunar landing, which is detailed in the section on the Crew Capsule.

5.1.4 Abort Options

The ERM supports abort options for the piloted mission. A successful abort results in the completion of the planned mission and/or ensures crew survival.

5.1.4.1 Descent Abort

The ERM is double engine-out failure tolerant (1 out of 3 engines operable) for landing; however, two engines must be operable to complete an abort to lunar orbit. An abort to lunar orbit (vs. an abort to the lunar surface with degraded performance) will be accomplished when the failure is such that a stay on the lunar surface is not desirable. Aborts to lunar orbit are available at any time during the landing sequence, and are initiated by jettisoning the LBM and igniting the primary propulsion system of the ERM to complete orbital injection.

5.1.4.2 Surface Abort

Following landing on the lunar surface, an immediate abort (i.e., accomplished within a matter of minutes) to lunar orbit can be initiated within the first 3 hours after touchdown. After 3 hours, the ERM is powered down and a maximum of 24 hours is required before an abort to lunar orbit can be completed. Aborts to lunar orbit are available at any time during the nominal 28-day mission stay.

5.1.4.3 Ascent Abort

During the ascent burn, the ERM is single engine-out capable (2 out of 3 engines operable). A double engine-out abort is possible only during the final phase of the insertion burn.

5.1.4.4 Trans-Earth Injection Abort

The ERM is double engine-out capable (1 out of 3 engines operable) during the Trans-Earth Injection (TEI) burn.

5.2 Stage Design

5.2.1 Configuration

The general configuration of the Earth Return Module is presented in Figure 5-6.

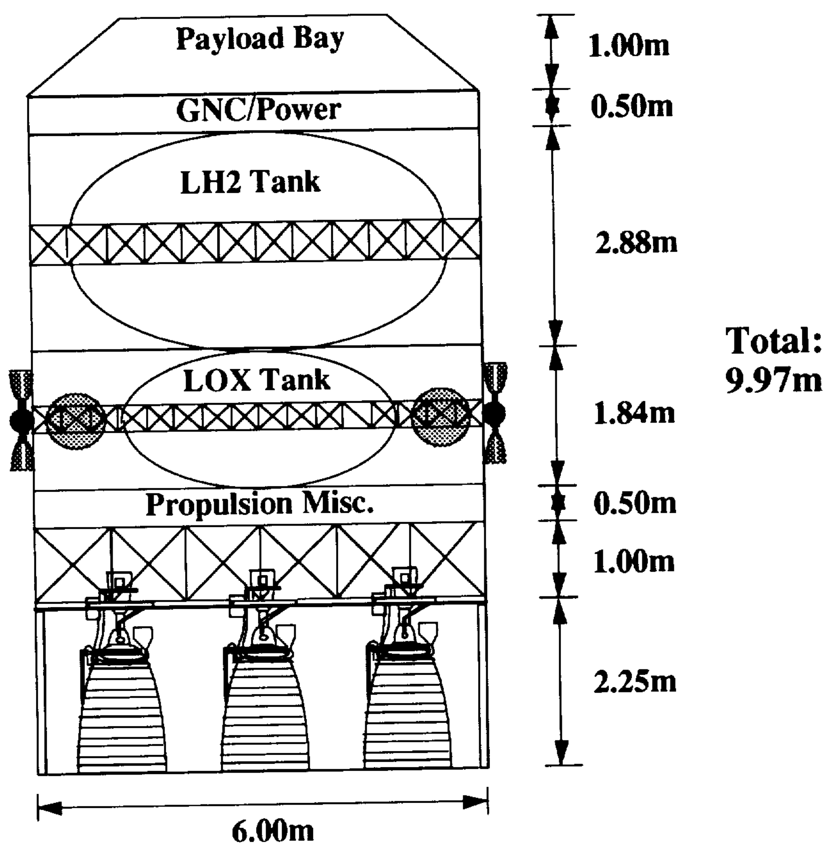


Figure 5-6

General Stage Configuration

This structure begins at the aft end of the ERM with space for the three RL-10 engines. This open space is surrounded by a thick structure/shroud that holds the ERM to the LBM and contains the lines for power and status. In addition, the space allows room for the main engines to gimbal the necessary amount (4°). This section of the ERM also contains the ERM-LBM interface and the landing gear setup, shown in Figure 5-7. At present, there are only two lines leading to the LBM: status and power. As stated above, these will be linked to the LBM by running the lines either through or on the edge of the connecting shroud. This interface will also contain explosive bolts that will be used when the LBM is staged. As for the landing gear, it is supported by four slots in the ERM. Each landing leg is about 4.2m long. Two meters of this length is slotted into the ERM, while the rest extends over into LBM slots immediately aft of the ERM ones. During landing operations, the landing gear deploy by sliding down about 1m and locking, suspending the bottom of the ERM about 2-3m off of the ground. More information on the landing gear can be found in Section 5.3.1 on ERM structures.

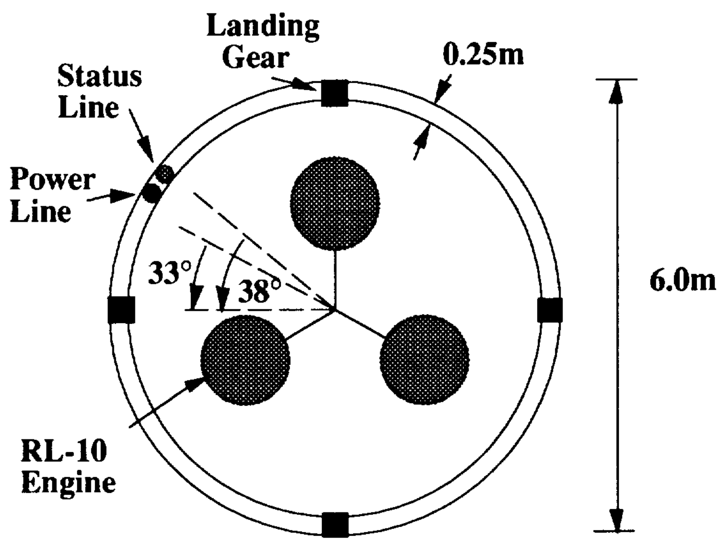


Figure 5-7
Aft Interface/Landing Gear

The main propulsion system, shown in detail in Figure 5-8, consists of three types of tanks and piping. The heaviest tank, the LOX tank, has been placed below the larger LH2 tank to increase the stability of the ERM during landing. Each tank is supported by an internal truss system, with the tanks of the RCS system supported by the same truss as the LOX tank. Lines run from the helium tanks to the LOX and LH2 tanks to pressurize them, and the LOX and LH2 lines run to fill valves and the combustion chambers of the main engines. It is important to note that the main outlet of the LH2 tank runs through the center of the LOX tank. This was designed to minimize propellant loss due to travel through pipes. An additional helium tank has been placed in the lower piping section of the ERM to pressurize the fuel entering the combustion chamber and facilitate stopping and restarting the engines. More details on the individual components can be found in the section on ERM propulsion. The piping schematic for the Reaction Control System has not been included with Figure 5-8 as it was deemed to be too unwieldy to fit into the current diagramming format. However, the two propellant tanks (MMH and N2O4) are located on either side of the LOX tank, with the pressurization tank (not shown) in the same position 90° from the other two tanks out of the page. The RCS engine combination consists of sixteen engines. These are arranged in clusters of four with one cluster in each cardinal direction. This gives the ERM as much maneuverability and redundancy as possible. A complete schematic of the RCS system can be also be found in the section concerning ERM propulsion. Each tank that requires cryogenic storage has also been insulated. This insulation is, on the average, 20cm thick and is wrapped on the outside of every cryogenic tank and pipe.

The engines are mounted 0.55m off the centerline, and at an angle 6.5° off the cylindrical axis. The engines are off axis to thrust through the vehicle's center of mass in case one or more engines fail. The 6.5° parameter degrades performance by cos(6.5°) to 99.4% of ideal. If all engines are thrusting, each is gimbaled -4° to 2.5° off of the cylindrical axis. The nominal 2.5° off degrades performance to 99.9% of ideal.

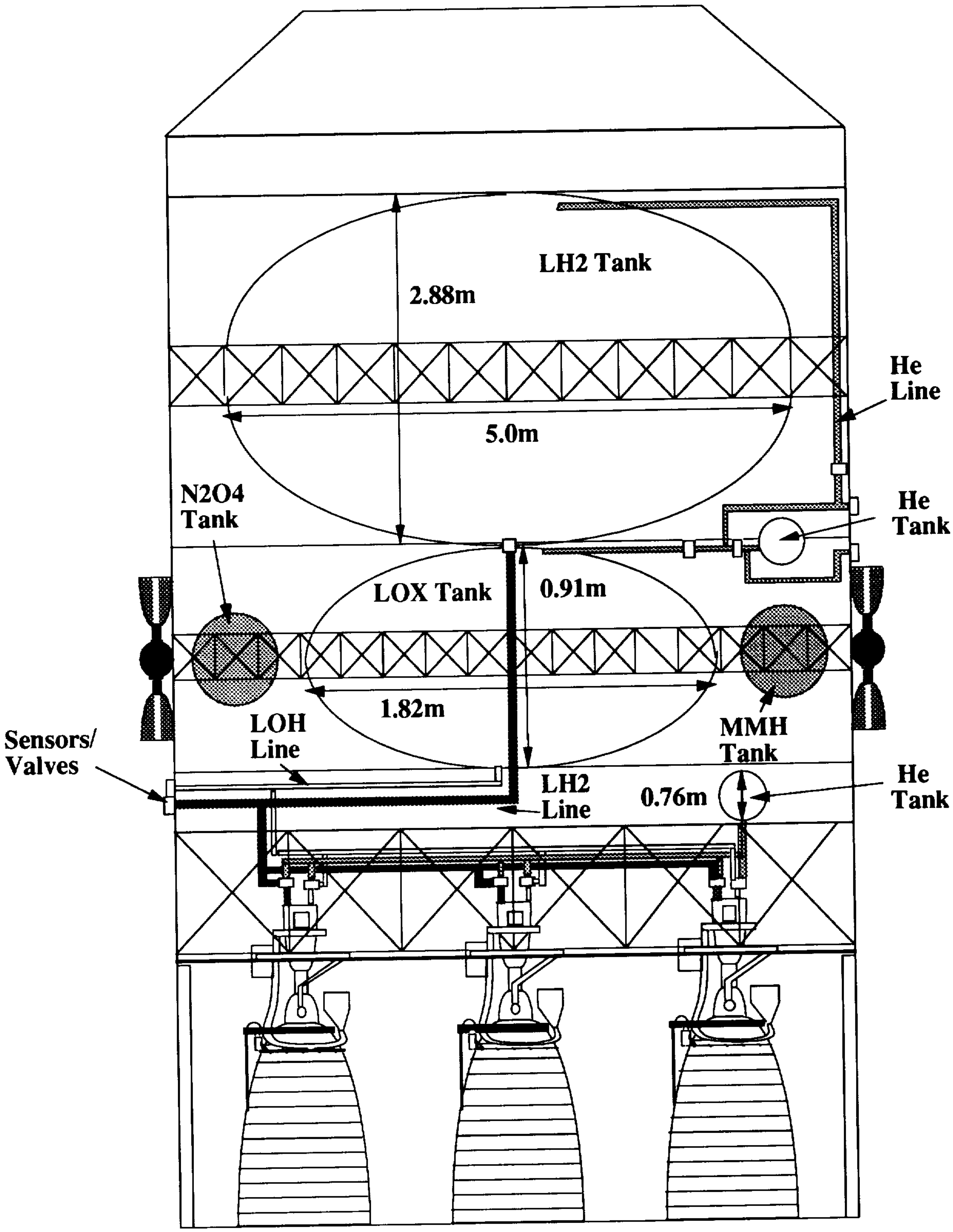


Figure 5-8
Main Propulsion Configuration

Following the propulsion system on the ERM comes the space alloted for the power system and GNC. This is depicted in Figure 5-9. Of the GNC components, the radar altimeter and the antenna beacons are arranged around the outside edge of the ERM because their functions involve interaction with the outside environment. The Star Trackers will be located at all four cardinal directions, like the RCS, and the Sun Sensors will be located as shown in Figure 5-9. The power system is located in the center of the ERM , and is

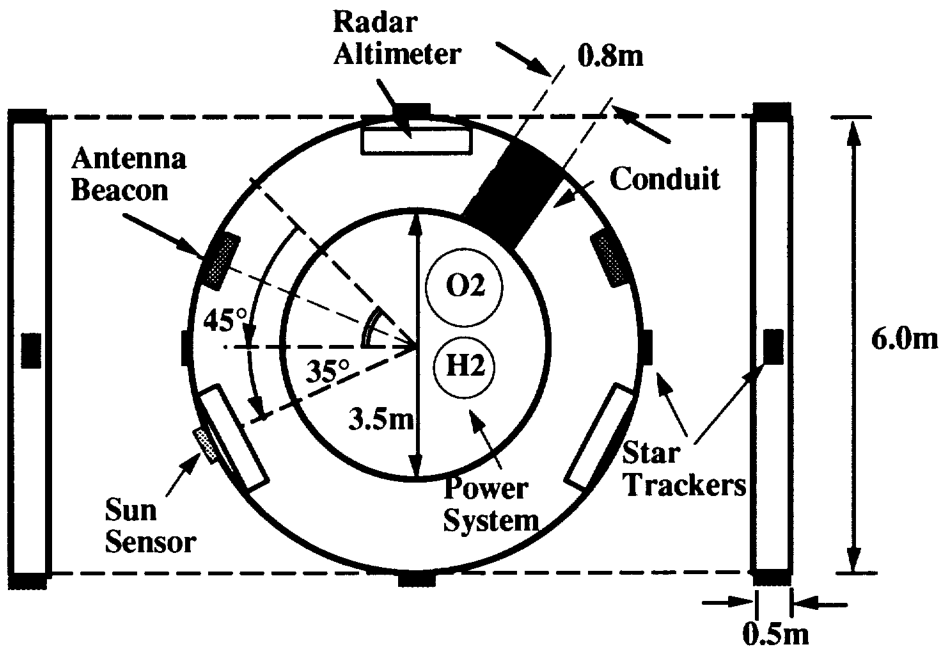


Figure 5-9
Power/GNC Configuration

composed of a H2 tank, an O2 tank, and a catalyst chamber. Running from the edge of the ERM to the power system is conduit provided for maintenance purposes. The power system is connected to all other systems by a series a power lines that will be run along the outer wall of the ERM. More detail on the power cells can be found in the section concerning ERM power.

After the Power/GNC section of the ERM is the payload bay and the forward interface. Depicted in Figure 5-10, the payload bay consists of a open area with a support column in the middle. This open area has been designed to hold a maximum capacity of 27 cubic meters and a maximum mass of 2300 kg. A sliding door has been provided to load and unload the payload. Opposite the door are the power, status, and GNC lines that interface with the CM. These are shielded by a shroud that overlaps the CM-ERM interface. Finally, there exists the physical connection to the ERM, which is composed of a series of clamps and is terminated by four explosive bolts.

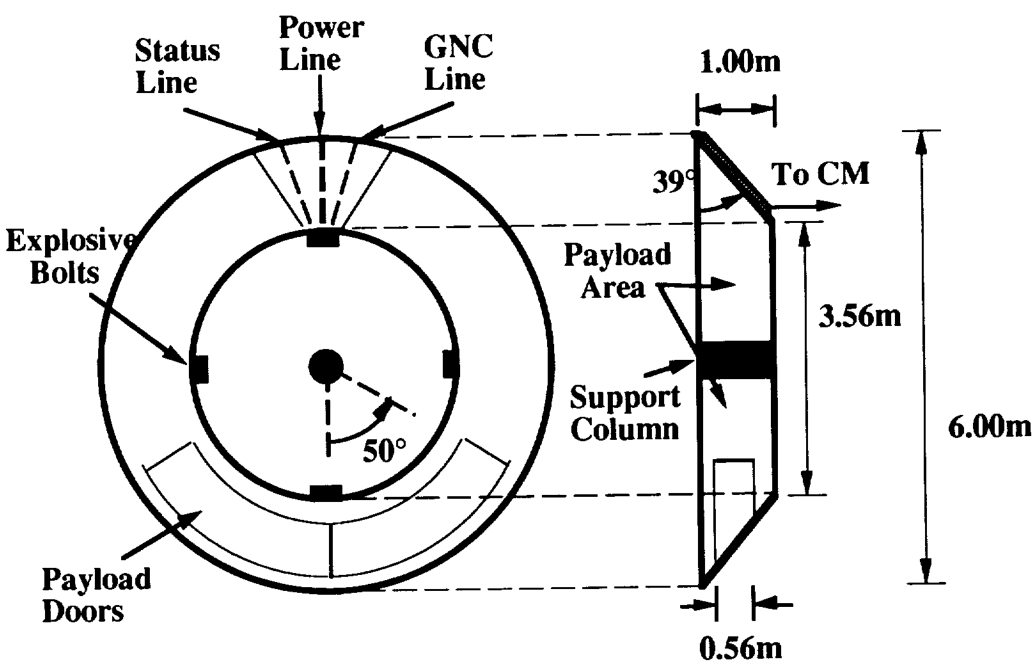


Figure 5-10
Payload Bay Configuration

5.2.2 Vehicle Interfaces

This section documents the interfaces for the ERM.

5.2.2.1 Lunar Braking Module

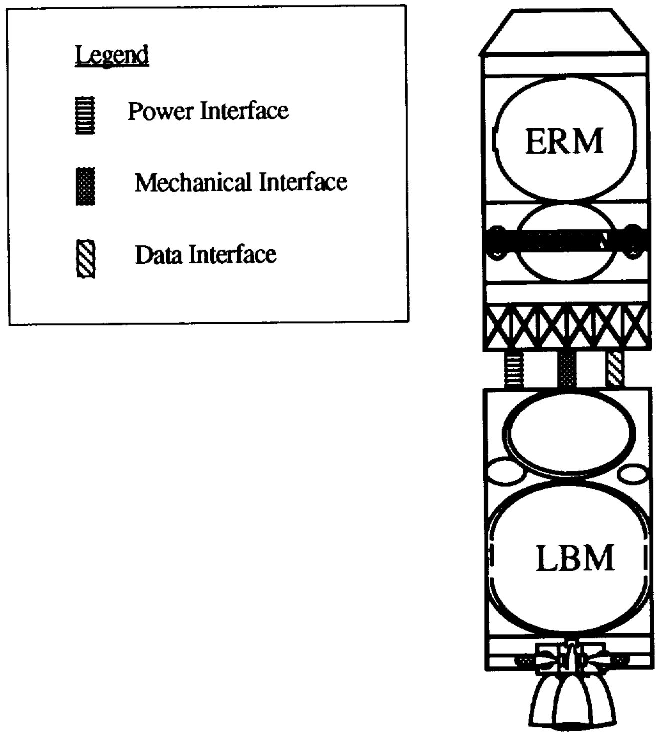


Figure 5-11
ERM/LBM Interface

5.2.2.1.1 Mechanical Interface

The mechanical interface between the LBM and ERM consists of explosive bolts for stage separation.

5.2.2.1.2 Data Interface

The data interface is a database between the LBM and ERM which transmits LBM status to the computers in the CM.

5.2.2.1.3 Power Interface

The power interface between the LBM and ERM connects the fuel cells in the ERM to all subsystems in the LBM.

5.2.2.2 Crew Module

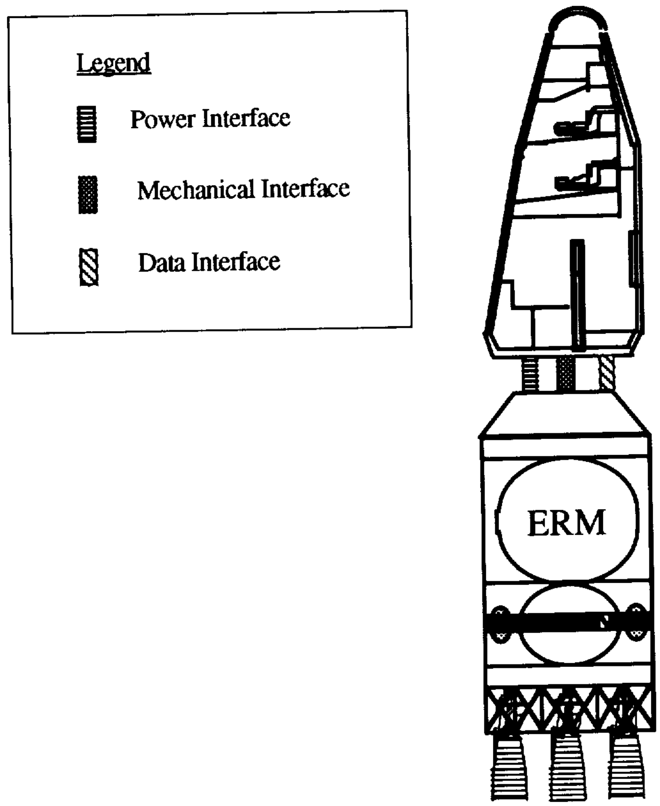


Figure 5-12
ERM/CM Interface

5.2.2.2.1 Mechanical Interface

The mechanical interface between the ERM and CM consists of explosive bolts for stage separation.

5.2.2.2.2 Data Interface

The data interface between the ERM and CM will transmit information to the CM which monitors the status of all other stages and transmit data for GNC.

5.2.2.2.3 Power Interface

The power interface between the ERM and CM will provide power for the CM from the fuel cells in the ERM.

5.3 Subsystem Design

5.3.1. Structural Design

A summary of the ERM structural design is included here as a service to the reader. For those interested in the methods of the design or desire to understand the structural trade-off that were involved in the choice of this structural configuration you are referred to section 2.1.4 and 2.2.1 in Volume II.

The geometrical description and masses are summarized below in Tables 5-4, 5-5, and 5-6.

Table 5-4: ERM Hydrogen Tank Design Parameters

Hydrogen Tank	
Hydrogen Mass	2723.08
Hydrogen Volume	38.35
Hydrogen Tank Volume	40.27
Hydrogen Tank Radius	2.50
Hydrogen Tank Cap Radius	1.25
Hydrogen Tank Cap Volume	32.72
Hydrogen Tank Main Volume	7.55
Hydrogen Tank Main Height	0.38
Hydrogen Tank Cap Eccentricity	0.87
Hydrogen Tank Cap Area	45.75
Hydrogen Tank Body Area	6.04
Hydrogen Tank Area	51.79
Hydrogen Tank Wall Thickness	0.0009
Hydrogen Tank Structure Mass	70.18
Hydrogen Tank Coating Thickness	0.0010
Hydrogen Tank Coating Mass	393.61
Hydrogen Tank Height	2.88
Hydrogen Tank Insulation Mass	559.34
Hydrogen Tank Mass	463.78

Table 5-5: ERM Oxygen Tank Design Parameters

Oxygen Tank	
Oxygen Mass	14976.92
Oxygen Volume	12.18
Oxygen Tank Volume	12.79
Oxygen Tank Radius	1.82
Oxygen Tank Cap Radius	0.91
Oxygen Tank Cap Volume	12.63
Oxygen Tank Main Volume	0.16
Oxygen Tank Main Height	0.02
Oxygen Tank Cap Eccentricity	0.87
Oxygen Tank Cap Area	24.25
Oxygen Tank Body Area	0.17
Oxygen Tank Area	24.42
Oxygen Tank Wall Thickness	0.0008
Oxygen Tank Structure Mass	28.98
Oxygen Tank Coating Thickness	0.0010
Oxygen Tank Coating Mass	185.62
Oxygen Tank Height	1.84
Oxygen Tank Insulation Mass	263.77
Oxygen Tank Mass	214.59

Table 5-6: ERM Configuration Summary

Configuration	
Stage Radius	3
Total Height	8.97
Insulation Mass	823
Casing Mass	1952
Rocket Truss Mass	267
Tank Mass	678
Landing Legs	1200
Structural Mass	4097
Engine Mass	501
Stage Dry Mass	5421
Stage Wet Mass	23121
Vehicle Wet Mass	28821
Structural Mass Fraction	18%
Structural Fuel Fraction	23.1%

5.3.2 Propulsion

5.3.2.1 Primary Propulsion System

The primary propulsion system of ERM stage is shown in Figure 5-13 on the following page. It consists of three RL10A-4 engines rated at 92,518 N nominal thrust and operating each at a 5.5:1 mixture ratio of oxidizer to fuel. The net positive suction head (NPSH) required by the engine turbopumps is provided by pressurizing the vehicle propellant tanks with helium gas at 272 atm. Propellants are delivered to the main engine turbopumps through feed ducts from the vehicle propellant tanks. The feed ducts contain flex joints to accomodate engine gimbaling and are overwrapped with a three-layer, double aluminized Kapton radiation shield.

The primary propulsion engines run on a bipropellant combination of liquid oxigen oxidizer and liquid hydrogen fuel. Both propellant tanks are cylindrical with semi-spherical endcaps, and are constructed of a thin steel core overwrapped with pre-stressed graphite composite fibers and a 20 cm layer of aluminized Kapton insulation. The oxidizer tank is 1.84 m tall and 3.64 m in diameter; the fuel tank is 2.88 m tall and 5 m in diameter.

Pneumatically actuated prevalves located at the propellant tank outlets provide series redundant backup for the engine inlet shuttof valves. A parallel set of pyro valves and solenoid valves upstream of the pneumatic actuation control solenoid valves provides two-failure tolerance against inadvertent opening of the engine inlet shutoff valves. The pyro valves will be fired open after the ERM stage is deployed a safe distance from the LBM stage. The system also has manual fill and drain valves to load propellant and pressurant gas into the system, as well as additional manual valves for system leak checking on both sides of the pyro-isolation valves and regulators. Check valves insure that the fuel and oxidizer can never mix anywhere in the system, except in the engine. Finally, pressure transducers, filters, temperature sensors, and line and component heaters are provided to ensure proper subsystem operation. A mass distribution of the entire propulsion system is given in Table 5-7.

Table 5-7: Mass Distribution of LBM Primary Propulsion System

COMPONENT	MASS [kg]
Empty Fuel Tank	463
Fuel Mass	2,707
Empty Oxidizer Tanks	214
Oxidizer Mass	14,949
Empty Helium Tanks	35
Helium Mass	32
Monitoring equipment	20 (estimated)
Propellant lines	26 (estimated)
Valves	39
Engine mass (3 RL10A-4 engines)	504
TOTAL FUELED WEIGHT	18,989 kg

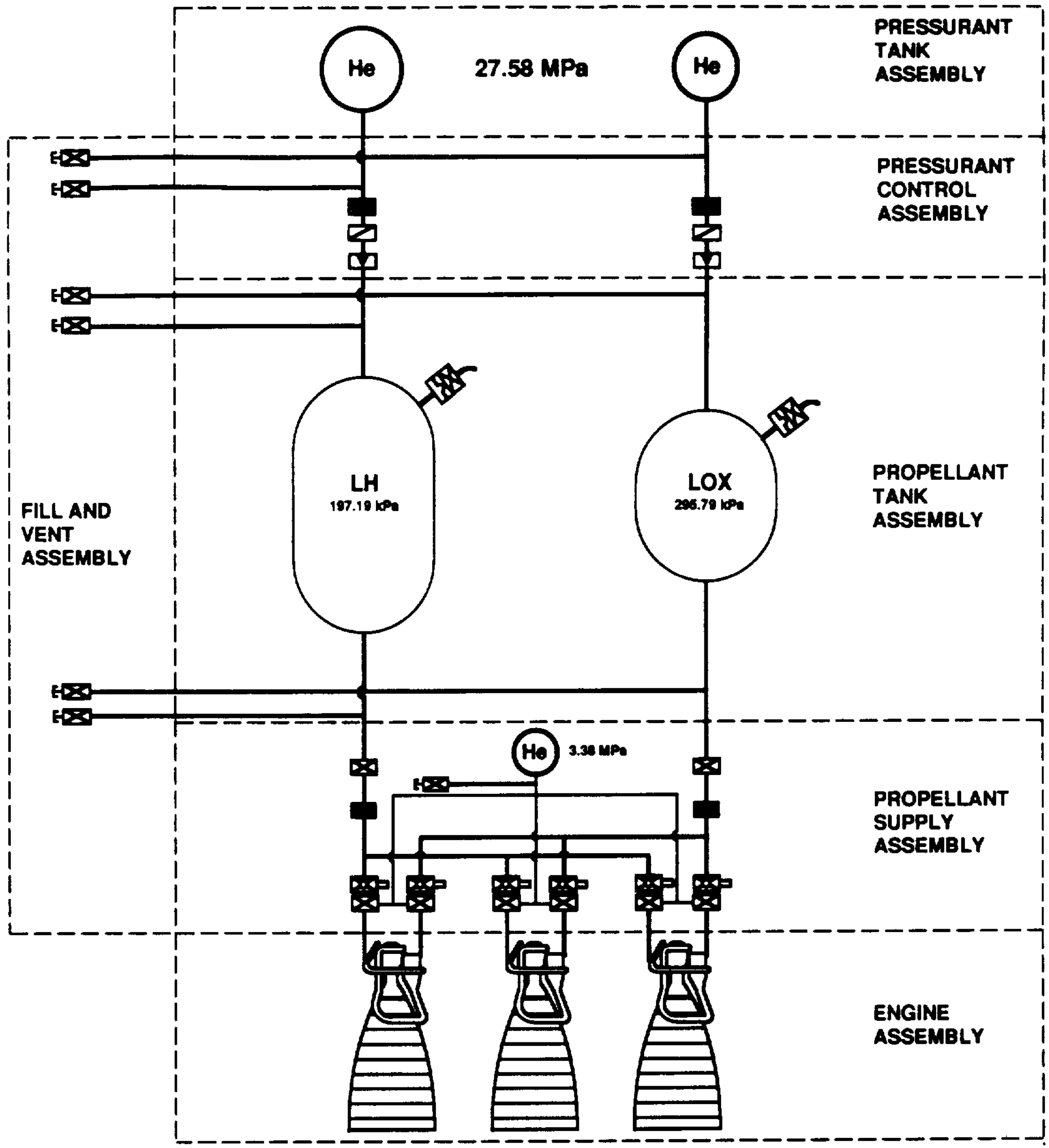


Figure 5-13
ERM Primary Propulsion System

5.3.2.2 Reaction Control System

The reaction control system of the LBM stage consists of two redundant subsystems configured as shown in Figure 5-14. Each subsystem consists of 8 R-4D thrusters operating on a 1.65 mixture ratio of oxidizer to fuel and fed by two propellant tanks. The thrusters are divided into quadruple clusters which are placed along the periphery of the spacecraft, making a total of 16 thrusters and four propellant tanks for the complete system.

The system utilizes a bipropellant combination of nitrogen tetroxide oxidizer and monometylhydrazine fuel. The propellants are stored in separate spherical tanks of identical size; each tank is 1.96 m in diameter. Both tanks are constructed of a thin steel core overwrapped with prestrssed graphite composite fibers; no thermal insulation material is required. Propellants are equipped with a Teflon diaphragm positive expulsion device which insures eficient tank evacuation.

A pressurant tank stores helium at about 272 atm, and a quad redundant regulator — coupled with a burst disk and relief valve— regulates flow. Together, they insure a 15 atm feed pressure to the propellant tanks, even after any single regulator failure. There are burst disks and pyrotechnically actuated squib valves to isolate propellants from the engine (and high pressure gas from the propellant tanks) until the system is ready for operation. This system also has manual fill and drain valves to load propellant and pressurant gas into the system, as well as additional manual valves for system leak checking on both sides of the pyro-isolation valves and regulators. Check valves insure that the fuel and oxidizer can never mix anywhere in the system, except in the engine. Finally, pressure transducers, filters, temperature sensors, and line and component heaters are provided to ensure proper subsystem operation. A mass distribution of reaction control system components is given in Table 5-8.

Table 5-8: Mass Distribution of the ERM Secondary Propulsion System

COMPONENT	MASS [kg]
Empty Fuel Tanks	42
Fuel Mass	944
Empty Oxidizer Tanks	42
Oxidizer Mass	1557
Empty Helium Tanks	11
Helium Mass	20
Monitoring equipment	20 (estimated)
Propellant lines	26 (estimated)
Valves	62
Engine mass (16 R4-D engines)	60
TOTAL FUELED WEIGHT	2784 kg

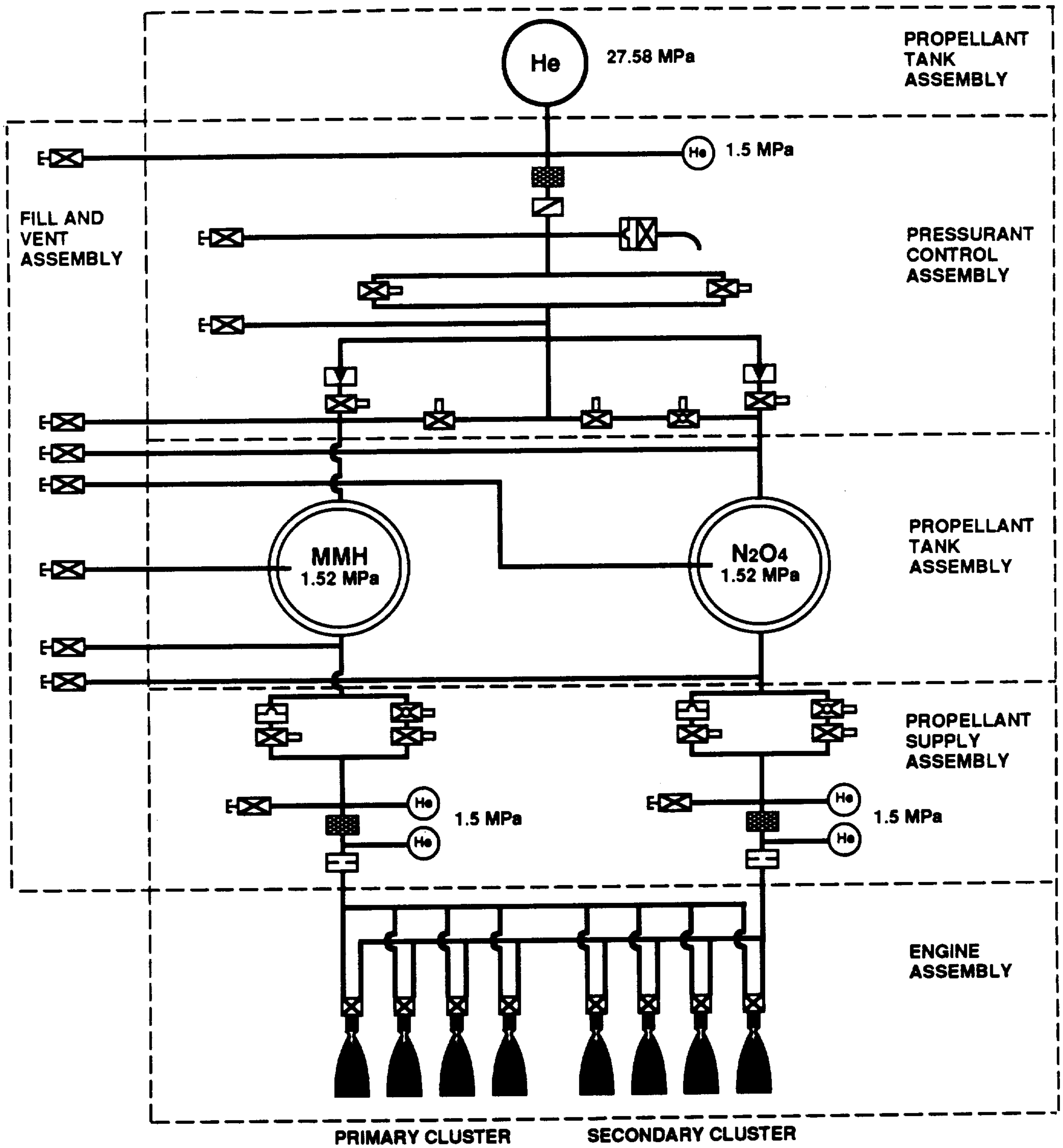


Figure 5-14

ERM Secondary Propulsion System

5.3.3 Power and Thermal Control

5.3.3.1 ERM Power Supply

The ERM will be the primary power stage of the upper three stages of the Piloted Vehicle. The power-time profile of the ERM calls for 6062 W for the first 3.1 days. Then while the ERM is on the lunar surface for 28 days it will use about 1000 W of power. The latter part of the profile has the ERM use 6062 W for another 3.1 days. Therefore the total power consumption for the ERM module comes out to be approximately 6062 W for 6.2 days.

The ERM must also supply power to the LBM and the CM during much of the mission (Note the additional power requirements in each of those sections).

The given power consumption of the ERM can be supplied by placing 306 kg of reactant mass in the ERM reactant tanks and 33.9 kg worth of fuel cell mass in the ERM vehicle. The reactant breaks down as 272.5 kg O_2 and 34.1 kg H_2, or as 0.239 m^3 of O_2 and 0.480 m^3 of H_2.

The ERM has a pair of spherical fuel cell reactant tanks which are separate from the propellant storage tanks. The reactants are cryogenically stored at 690000 Pa (100 psia) which is the minimum input pressure for the fuel cells and pumped out of the tank via a Helium gas feed system. Storage of the above volumes of reactants requires a LOX tank of radius 0.385 m, dry mass of 28 kg, and an LH_2 tank of radius 0.490 m, dry mass of 46 kg.

5.3.3.2 ERM Thermal Control

The primary thermal control concerns on the propulsion stages are the cryogenic storage systems, the RL-10 engines, and the stage interior. The RL-10's are regeneratively cooled and have maximum rated burn times; therefore it is not necessary to provide an additional thermal control system for the engines. Thermal control of the stage interior is maintained passively through the applications of a reflective outer coating of silverized aluminum. The cryogenic systems are thoroughly described in section 6.3.

The spherical cryogenic reactant tanks for the ERM fuel cell system are wrapped in layers of Mylar insulation to hold a .175% boiloff rate over 34 days. The LH2 tank receives 268 layers, while the LOX tank receives 133 layers, and the total insulation mass is 51.78

kg.Insulation for the ERM stage is designed to allow 0.175% fuel mass boiloff over a period of 34 days.

The radius of the hydrogen tank outer surface are set at 2.5 m for a height of 0.38 m, with ellipsoidal endcaps of minor axis length 1.25 m. The oxygen tank is an ellipsoid of major axis radii measuring 1.82 m and minor axis radius 0.92 m.

Two hundred and forty-six layers of aluminized mylar are required to insulate the hydrogen tank, representing a total thickness of 17.65 cm, while the oxygen tank requires only 133 layers totalling 9.54 cm thickness. The total mass of the insulation is 526.75 kilograms.

5.3.4 Sensors in Earth Return Module

In order to minimize the components in the crew module (CM), the four star trackers and sun sensor will be placed in the Earth Return Module (ERM). For redundancy, three star trackers will be used, with one sun sensor to provide coarse spacecraft alignment. A more detailed discussion of the selection process for the components is in Volume II, Chapter 5.

5.3.4.1 Location of Sensors

The four CT-601 Solid State Star Trackers must be positioned on the ERM so that star trackers can locate the catalog stars stored in the ephemeris. The sun sensor must be positioned so that it faces the sun. The star trackers must also be shielded from the sun, as in the alignment shown in ???(Figure 5-x+3)???. This set up means that the spacecraft will have to fly "upside-down" for either the trip to or return from the moon in order to keep the star trackers pointed away from the sun. Because the spacecraft must be rolled for thermal control, at most, only two of the star trackers will be operating at a single time. The only additional requirement on the ERM is that an interface exist between the star trackers and the main guidance computer in the CM, so that the star tracker updates may be used.

5.3.4.2 Radar Altimeters

There are three radar altimeters. These are placed 120° apart around the surface of the ERM, sunk into the walls. These three measurement devices provide range data in the vicinity of the lunar surface. Using doppler shift techniques, it is also possible to obtain surface closing velocity. All three used together can increase accuracy, though each individual component is designed to provide adequate accuracy in isolation.

5.3.4.3 Antenna Beacons

There are three antenna beacons. They are placed in the guidance area of the module. These units are connected to receiver and transmitter antennae located in the communications area which allow the antenna beacons to send/receive signals.

5.3.5 Communications and Control System

The high gain antennas and pointing system for the piloted mission communications system are located on the ERM due to a limited space on the CM. The high gain system will not be used during reentry, so it will not be missed when the ERM is ejected from the CM. Details on the communication system are described in Volume II sections 4.2.3 and 4.3.2.

5.3.6 Status

The ERM again is mainly a propulsive stage and resembles the PTLI. Here the issue of abort is nonsensical. The ERM must be patched, glued, or wired together to withstand the return to earth.

5.3.7 Subsystem Interfaces

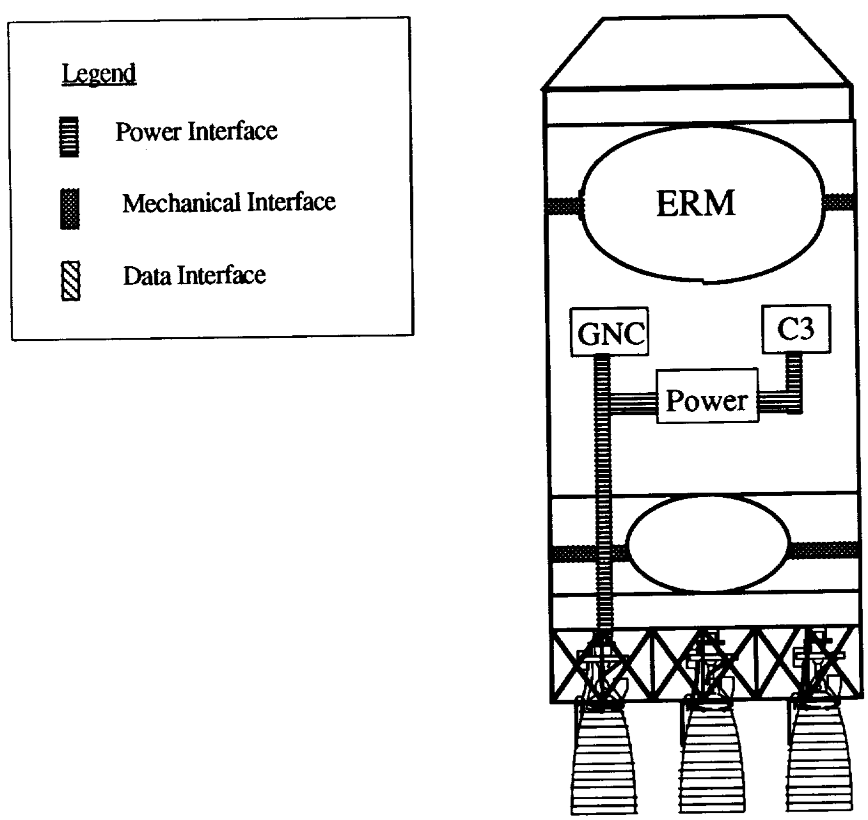


Figure 5-15
ERM Subsystem Interfaces

5.3.7.1 Mechanical Interface

The mechanical interfaces consist of structural trusses which attach the tanks to outer shell.

5.3.7.2 Power Interface

The power interface supplies power from the fuel cells to propulsion subsystems, GNC subsystems, and C3 subsystems.

6 Crew Module

The crew members of Project Columbiad will embark upon their journey to the Moon in a single, reusable capsule. This capsule, known as the Crew Module (CM), contains all crew systems; command, communication, and control systems; guidance navigation and control systems; as well as a sealed fuel cell capable of powering all onboard systems during the last two hours of reentry and landing. The CM has a biconic design which employs deployable wings, drogue chutes, and a parafoil to land the astronauts safely on a runway.

The CM has fully redundant systems to support various abort modes at many points during Project Columbiad's mission. All operations within the CM are monitored at all times by Status with all information fully displayed to both the crew members and the ground-based mission control. Landings, both on the Moon and the Earth, are preprogrammed with the capability for fly-by-wire over-ride for land point selection by the crew members.

6.1 Module Budget

Weight drives the design of any space vehicle, be it inhabited or uninhabited. Reduction of weight increases deliverable payload and decreases required propellant, structural mass, and launch and operation costs. Unfortunately, reducing weight also means reducing the amount of equipment carried aboard the spacecraft--creating the paradox of all spacecraft budget analysis:

- If you increase a vehicle's weight you decrease its payload and range, but
- If you decrease it's weight you reduce its mission capabilities and duration.

The Project Columbiad Systems integration team came up with a final CM mass limit of 5600 kg, a very tight restriction which the Crew Module had stay under while including all life support, communications, navigation, power and structural mass. In order to find an equitable tradeoff between mass reductions and included equipment, every subsystem on-board the vehicle followed a tight budget--bringing only what was absolutely necessary to satisfy all mission requirements safely and reliably. Only after repeated budget iterations was a budget found that satisfied all mission groups, from structures and propulsion to crew systems and guidance and control.

The following iteration procedure was used in the design of the capsule:

1. Inquire mass, power, and dimensions of all components (both standard and redundant) desired by each subsystem involved with the capsule.

2. Obtain the total power figure required and find the mass and volume of these power units from PTC.

3. Give this total mass and volume number to Structures to find total structural mass.

4. Check if total mass, volume, and power falls within requirements set by Systems Engineering. If it does not...

5. Look at mass, volume, and power for each group. Reduce this number and assign it as the budget for that group. Now the subsystems groups will start cutting out things that are not absolutely necessary.

6. Now return to number 1, repeating procedure until the final numbers satisfy Systems' requirements or cannot be lowered any further. If the final numbers cannot satisfy Systems' requirements, the requirements must be altered.

This process is even more exacting for this capsule design because of the aerodynamic requirements imposed upon the capsule. The CM's biconic design must capable of reentering the Earth's atmosphere at very high velocity, and landing on a runway like an airplane. This biconic shape enforces strict volume specifications, based on the diameters supported by the launch vehicles and the maximum weight that can be delivered by propulsion.

6.1.1 Final Crew Module Budget

After many iterations, a CM structural shell with the following dimensions was selected:

- External volume of 46.2 m^3.
- Overall length of 7.84 m.
- Maximum diameter of 3.56 m.

Within this shell, the final budgets for the individual subsystems on-board the CM are

Table 6-1: General Budget for CM

	WEIGHT	VOLUME	MAX POWER
Crew Systems	1155.2	16.79	2610
CCC	223.77	0.13	2182
GNC	35	0.08	175
Structures	2275	2.00	150
Power	21.6	0.04	N/A
Miscellaneous:	130.8	0.19	N/A
Abort Systems	378.04	2.12	0
Status	0	0	100
RCS	429.09	0.7	100
TOTAL	4648.50	22.05	5317
Margins	5578.20	26.46	5848.7

N/A: Not Applicable

Heat Shield has mass of732kg

The breakdown and layout of the general sections and individual subsystems are describe sections 6.2 through 6.8 of this chapter.

6.1.2 Budget Margins and Considerations

It is painfully obvious that all wiring, equipment racks, piping, cabinets and wasted volume cannot be completely enumerated. Also, some of the dimensions and volume estimates of the various components within the crew capsule are estimates from out-of-date textbooks, NASA design projects, and industry summaries. To account for these sources of error and to specify the size of a system which would operate in a spacecraft of this size, many of the subsystems have estimated the mass, power, and size requirements of various components. To compensate for these unavoidable errors, two things were done . First, all layouts have been drawn with three dimensional views (with cutaways) using Claris Cad™ to specify exact dimensions in metric units. Second, an error margin of +20% has been includes in the general CM budget. This gives a total mass of the CM (without the heat shield) of 5578 kg with the 20% margin, just shy of the 5600 kg limit imposed by Systems.

6.2 Crew Module Layout

Four drivers guided the layout of the Crew Module:

- •Required volume and geometry of the habitable crew station
- •Volumetric efficiency of the biconic design along its long axis
- •Structural requirements for the shell and doors needed for integrity
- •Geometric wing, body, and center of gravity requirements for successful aerodynamically assisted reentry.

The main objective behind designing the layout of the CM crewstations was satisfying the operational requirements of the various crew stations. The crew stations must be functional, yet uncluttered. This layout and operational style was achieved by breaking down the S/C functions into computer-aided tasks and monitorings. The use of computer tasking enables the pilots to act as supervisory managers rather than simple manual controllers. The on-board computer controls make S/C operation simpler and more efficient by creating easily assimilated graphic interfaces which do not overwhelm the pilots with excessive data and S/C complexities.

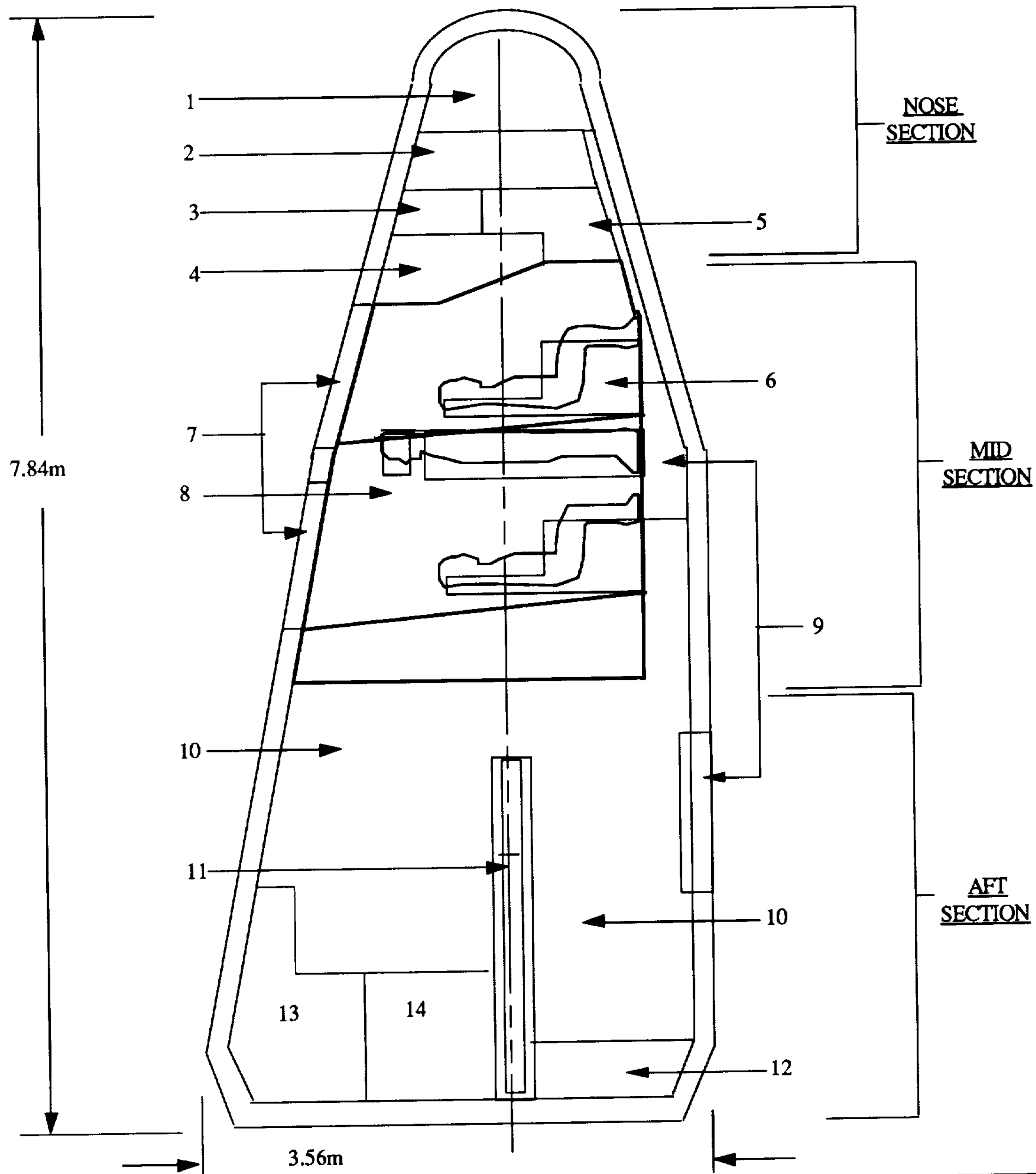
The habitable volume inside the CM was created with chair and component spacing to allow the crew to engage in all required activities. Pilots can walk between port and starboard seats, dress and undress, climb into hanging sleeping bags, and reach all intrumentation and controls easily. Adequate ceiling spacing exists to allow the crew to easily enter and leave the S/C during lunar landing with the aid of a pulley system which conducts the suited crew members along the 15 m climb from the doors to the surface.

The CM was designed to survive the launch, injection, braking, landing and reentry loads experienced through the mission. The adherence of the design to these structural considerations limited the amount of weight and space that can be dedicated to other systems within the CM. Therefore, it was important to minimize the weight and volumetric contribution of the supporting structure to include all other systems in the S/C design and meet the launch weight requirements. A semi-monocoque design was used as opposed to a monocoque design for the structure, because of its greater strength to weight ratio.

Finally, the mass centroid of the CM is at the same location as the design aerodynamic centroid for stable operation during reentry. The mass centroid calculation is provided in Section 6.2.4.

6.2.1 Basic Layout

The CM biconic shape is 7.84 m long and has a 3.56 m maximum diameter. The inner layout is divided into three general sections: the nose section, the mid section, and the aft section . These sections are shown in Figure 6-1 (a) and (b).



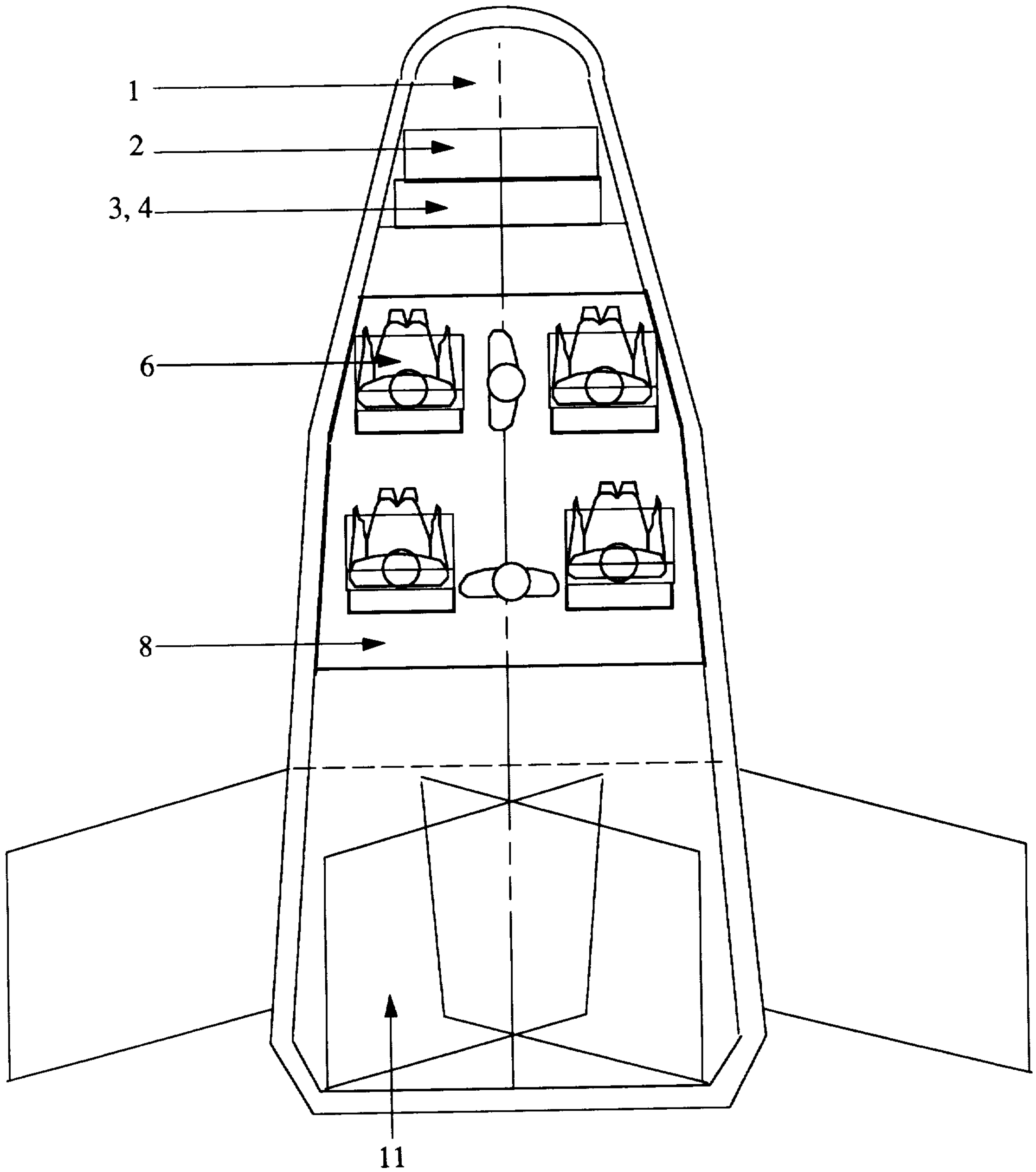
RIGHT SIDE VIEW 1 of 2

NOSE SECTION
1 RCS Engines and Fuel
2 Parasail Systems
3 GNC and CCC Systems
4 Instrument Panel
5 Crew Support Systems LiOH System

MID SECTION
6 Ejection Seat and Guide Rails
7 Escape/ Egress Hatches
8 Habitable Area
9 Landing Gear

AFT SECTION
10 Crew Support Systems
11 Wings and Deployment Systems
12 CM Reentry Power Systems
13 RCS Engines and Fuel
14 Drogue Chute System

Figure 6-1 (a)
Command Module Layout



TOP VIEW		2 of 2
NOSE SECTION	MID SECTION	AFT SECTION
1 RCS Engines and Fuel	6 Ejection Seat and Guide Rails	10 Crew Support Systems
2 Parasail Systems	7 Escape/ Egress Hatches	11 Wings and Deployment Systems
3 GNC and CCC Systems	8 Habitable Area	12 CM Reentry Power Systems
4 Instrument Panel	9 Landing Gear	13 RCS Engines and Fuel
5 Crew Support Systems LiOH System		14 Drogue Chute System

Figure 6-1 (b)

Command Module Layout

6.2.1.1 Nose Section

The main systems for guidance, navigation and control and command, control and communications are located in the nose section. These systems are supplied with power and thermal control facilities. Power is routed to the nose section from the main power interface with the ERM, which includes all power for the CM journey, except for the last two hours of Earth reentry. Data from the high-gain antennas and sun and star trackers located on the ERM is also transmitted through a connecting data bus. Pipes leading from the nose section convect heat away from the nose, maintaining a range of operational temperatures for the components. The parafoil system is also located in the nose section and is used in tandem with the drogue chutes during the descent to the Earth's surface. Also, the Negative Pitch RCS Engines and their N_2O_4 oxidizer, Mono-methyl Hydrazine fuel, and Helium pressurization tanks are located in the nose section.

6.2.1.2 Mid Section

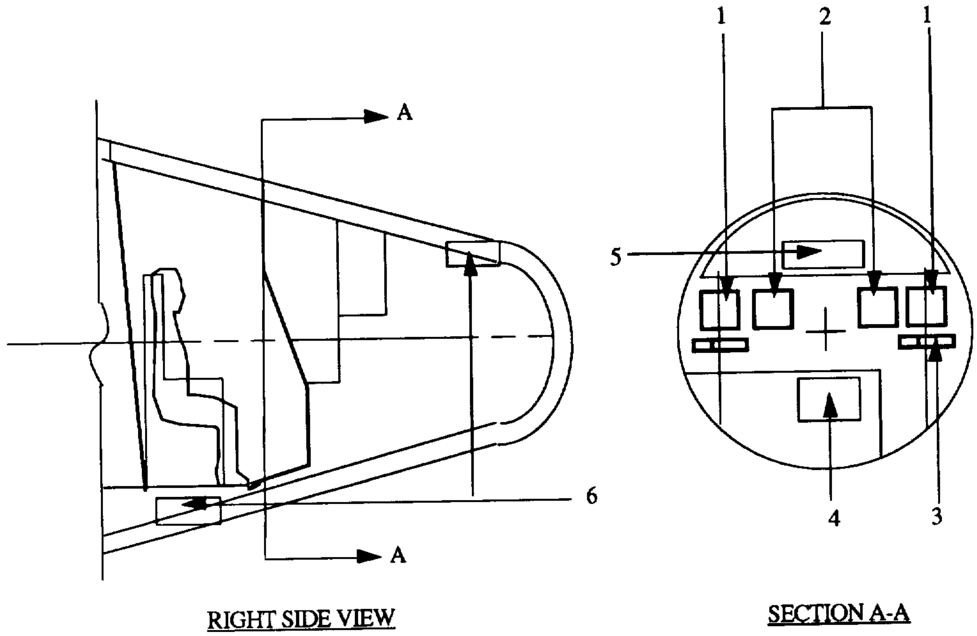
The Mid Section consists primarily of the habitable volume, the accessible crew support systems, and the crew monitored control panel.

6.2.1.2.1 Habitable Volume Systems

The habitable volume sections consist of the crew living area, including two sleepers, the four ejection seat units with guide rails, the main flight control panel, and the accessible crew support systems needed during the transits to and from the Moon. These accessible systems are located in the front, back, and on the base of the mid section. Also in the mid section are the two hatches which are blown off in the case of an abort. The rear hatch also serves as the main egress from the CM. This hatch system is discussed further in section 6.2.5.1.

6.2.1.2.2 Control Panels

Figure 6-2 shows a simplified view of the control panel. The layout of the control panel for the CM is greatly simplified by a computer display, and a joystick and keyboard system which provides a user-friendly approach to controlling and monitoring all the on-board systems. There are four main control panel sections: the left panel (mission commander's side), the right panel (the co-pilots side), the overhead panel, and the center console panel.



CONTROL PANEL DETAIL
1 CRT Displays
2 LCD Displays
3 Keyboard and Mouse-assisted Plotter
4 LiOH Cartridge Insert Area
5 Circuit Breakers
6 Video Cameras

Figure 6-2
Control Panel Layout

On the mission commander's side (left panel), there are two computer displays, a keyboard, a mouse/target plotting system, and a vertical control joystick for hovering. There are also the circuit breakers for powering the system up and down. All communications and GNC, video displays, warning displays, crew life support systems monitoring, heating control systems monitoring, wing deployment and retraction controls, and the controls for detaching the drogues are accessible to the mission commander through the mouse, keyboard and displays system provided. The computer display systems allow the mission commander to obtain useful information and status as well as suggested solutions to problems through a menu system.

The pilot's side (right panel) is identical to the mission commander's panel, allowing for all mission commander's responsibilities to be completed by the pilot at any time. This redundancy insures that no loss of control occurs due to the rest of the crew being inaccessible to the main controls and communication systems of the S/C.

On the overhead panel there are the lighting controls, the on board computer system instruments and the fuel purge controls.

The control panel operations are explained in detail in section 6.2.8.

6.2.1.2.3 Equipment Cabinets and Storage

Equipment cabinets provide space, power connections, and cooling to all on-board instruments and equipment on the CM. The cabinets also provide protection for the equipment against g-shocks and other severe loads. This is done by the use of rubber dampers at the connection points of the S/C to the cabinet.

Storage lockers similar to those used on the STS Orbiter are located in the rear of the mid section, and provide storage for the crew support systems needed for the lunar mission. These systems include the IVA suits, needed tools, equipment, food, utensils, and apparatus. The storage lockers are equipped with restraints as well as door latches needed to prevent the stored materials from forcing themselves out of the locker during CM maneuvers.

6.2.1.3 Aft Section

In the aft section there are the main crew support systems for the CM, the positive pitch, left and right yaw, left and right roll RCS Engines and fuel tanks, the drogue chute system, the on-board power system needed for the CM during reentry, the deployable wing systems. The rest of the power systems needed for the CM during the lunar mission are located in the ERM. Detailed layout of this section is located in section 6.2.2. of this volume.

6.2.2 Subsystems Layout

Each subsystem is layed out based on intra-system interfaces required by each subsystem. In addition these designs must efficiently fill all volume constrains, allowing adequate spacing for wiring, thermal piping, and securement to the CM. The added layout constraint of placing components in easily accessable locations has been imposed on all components which require crew maintainance and direct monitoring.

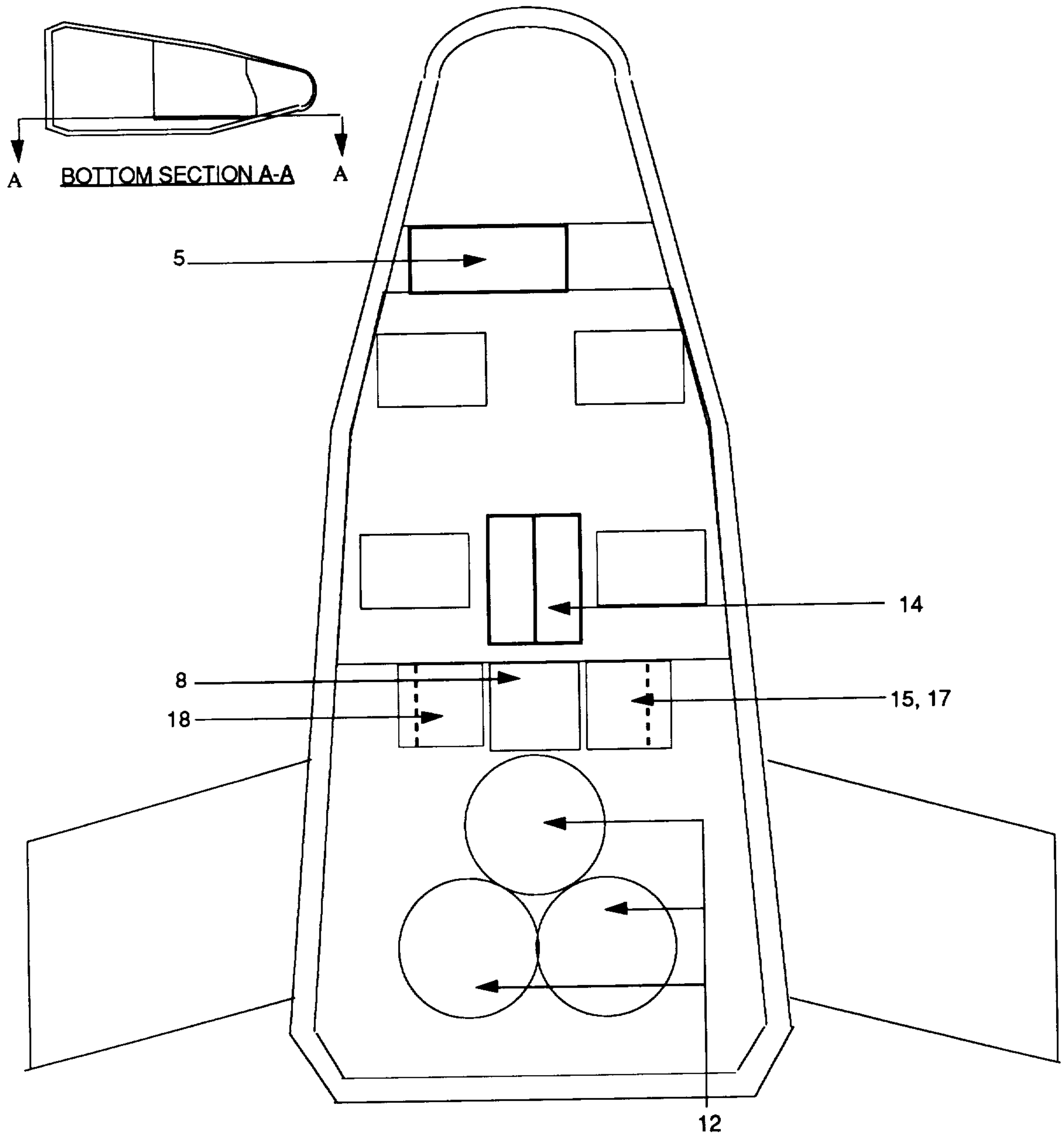
6.2.2.1 Crew Support Systems Layout in the CM

The crew systems components required the most ergonomic placement, due to their continuous interaction with life onboard the CM. Also, the atmosphere and thermal control systems must have open ports along the walls of the main cabin order to process the CM atmosphere. Crew Capsules has made other considerations in Crew Systems layout design to improve living conditions, making it easier to for the astronauts to perform all mission duties effectively. Such considerations include:

- Placing the hygiene station along side of the commode and near the biomedical/ first aid station.
- Placing the blowers of the thermal and atmospheric control systems behind the astonauts but far above the commode.
- Placing the lithium hydroxide spare cartridge door below the control panel, between the front two crew members.
- Placing hooks for the IVA helmets above the astronauts, and hooks for the IVA suites in the rear corridor of the main section.
- Providing easily accessible storage cabinets for all supplies, emergency equipment and scientific equipment.

Placing the hygiene station near the commode and biomedical station improves overall capsule hygiene and reduces proliferation of bacteria and odors. The location of the blowers creates circulation patterns which reduce spread of odors during use of the commode, and creates a gentle breeze which alleviates some symptoms of space sickness. Also, the sliding door operation of the commode increase habitable volume. The easily accessable location of the LiOH cartridges allows the mission commander or pilot to change the cartridges, when notified by the compuer on the control panel, without leaving their station. The IVA helmet hooks allow the astronauts to remove their IVA helmets for increased visibility and comfort, but keeps the helmets quickly accessable in case of emergency. The IVA suit hooks provide a place for the suits during sleeping and commode use. The cabinets provide easily accessable food, clothing, and shoe storage, as well as quit access to emergency safety balls and the IVA backpacks.

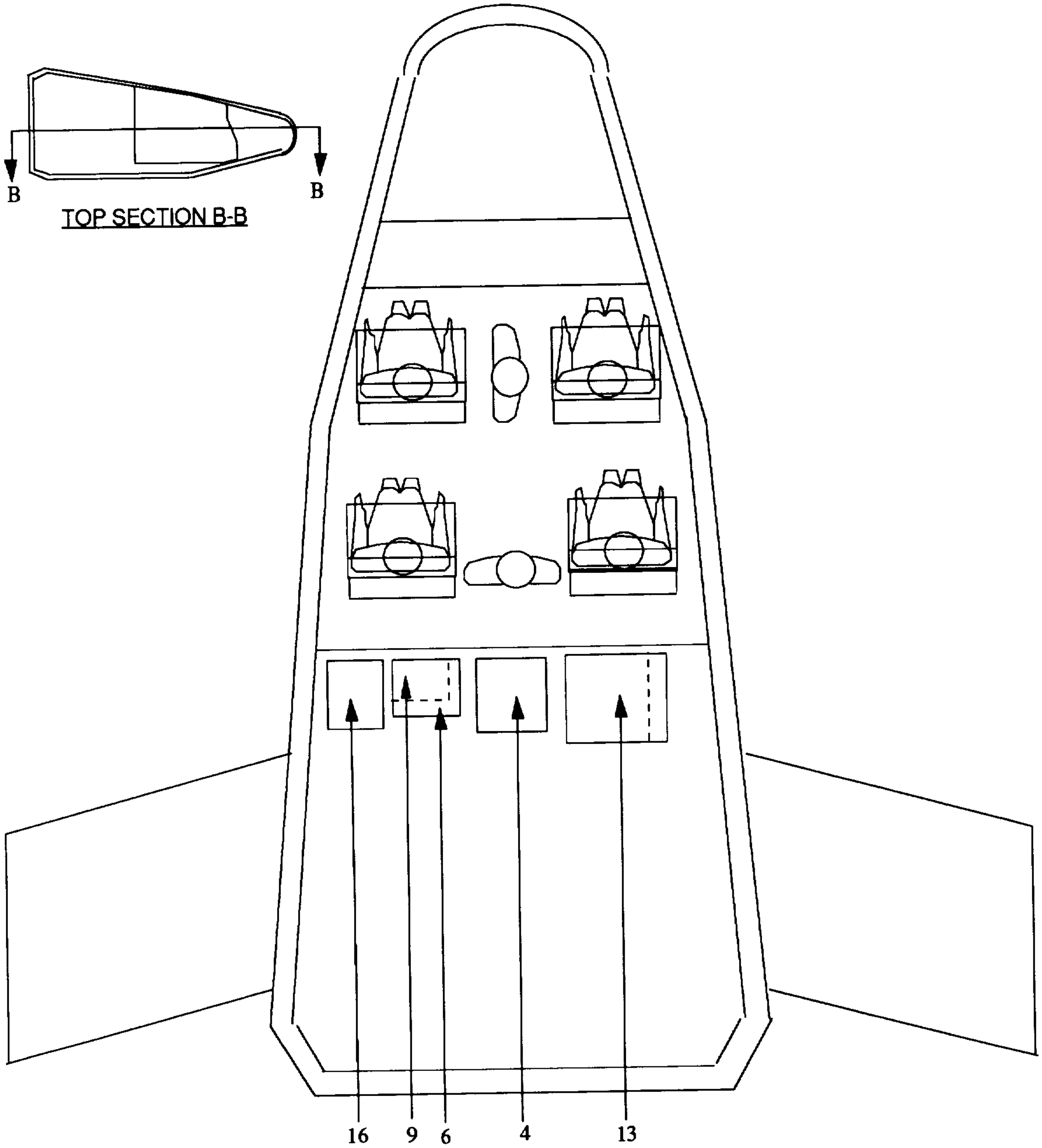
These features are demonstrated in the multi-layered, cross-sectional views of the Crew Systems layout in Figure 6-3 (a) through (g).



BOTTOM LEVEL TOP VIEW 1 of 7

1 Humidity Control	8 Commode	15 Toiletries
2 Thermal Control System	9 Hygiene Station	16 Tools and Cleaning Equipment
3 Mass Spectrometer	10 Water Tank	17 Extra Shoes and Clothing
4 Two Gas Bread-board Control	11 Nitrogen Tank	18 Over-garments
5 Lithium Hydroxide System	12 Oxygen Tank	19 Rescue Ball
6 Biomedical Station	13 IVA Backpacks	20 Sleeper
7 Water Pump	14 Food	21 Fires Suppression and Detection

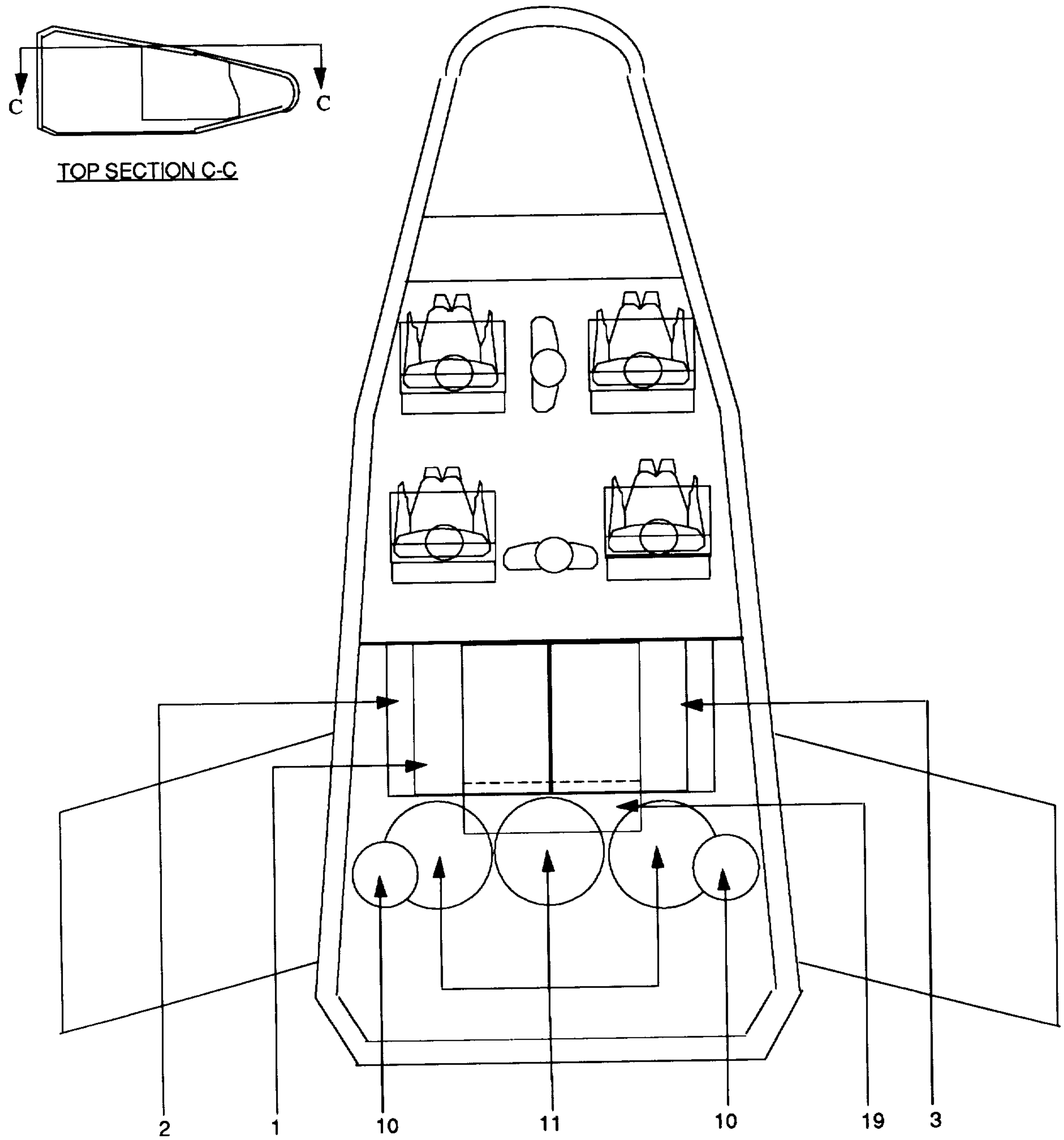
Figure 6-3 (a)



MIDDLE LEVEL TOP VIEW 2 of 7

1 Humidity Control	8 Commode	15 Toiletries
2 Thermal Control System	9 Hygiene Station	16 Tools and Cleaning Equipment
3 Mass Spectrometer	10 Water Tank	17 Extra Shoes and Clothing
4 Two Gas Bread-board Control	11 Nitrogen Tank	18 Over-garments
5 Lithium Hydroxide System	12 Oxygen Tank	19 Rescue Ball
6 Biomedical Station	13 IVA Backpacks	20 Sleeper
7 Water Pump	14 Food	21 Fires Suppression and Detection

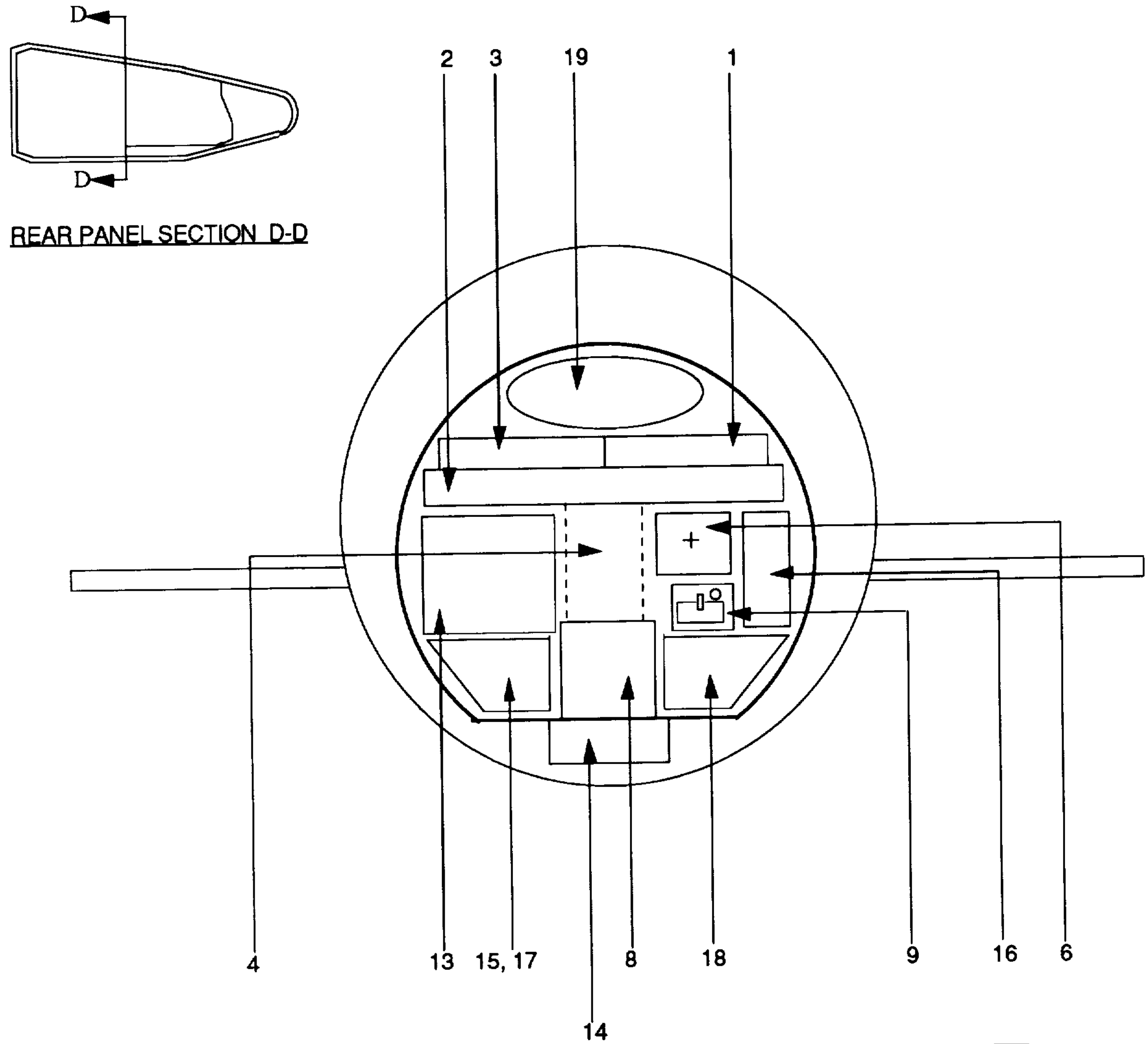
Figure 6-3 (b)



TOP LEVEL TOP VIEW 3 of 7

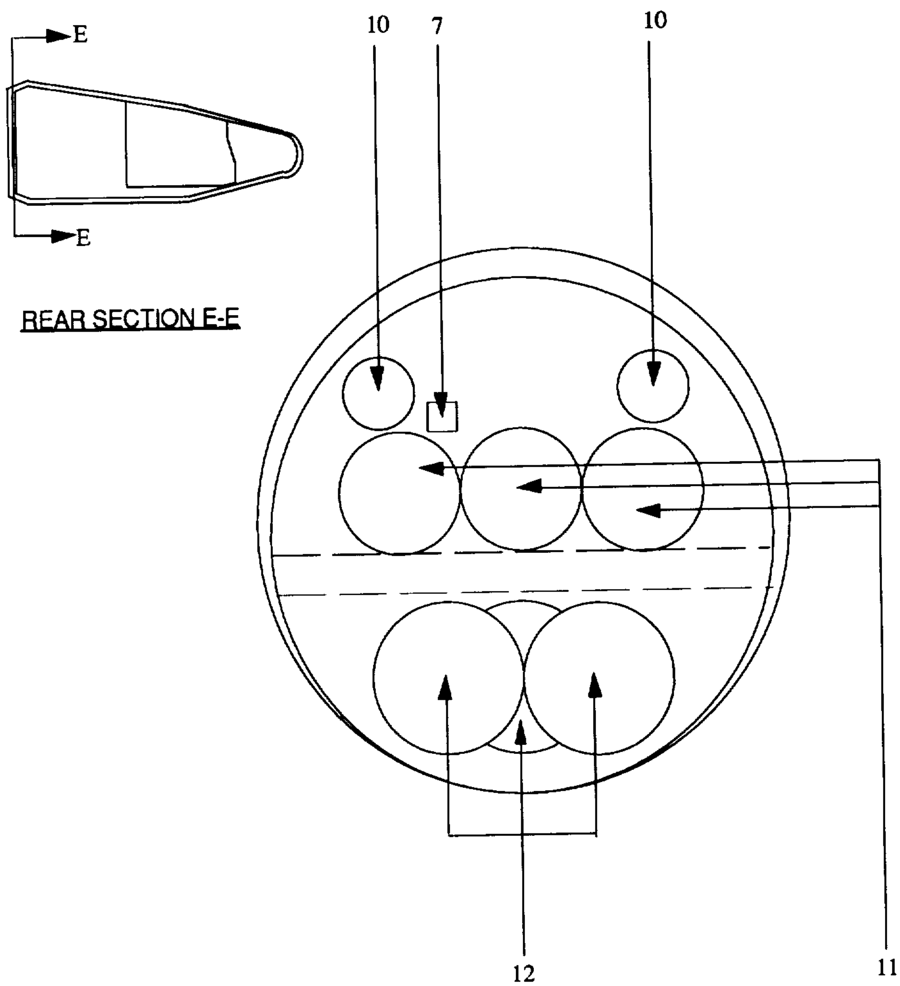
1 Humidity Control	8 Commode	15 Toiletries
2 Thermal Control System	9 Hygiene Station	16 Tools and Cleaning Equipment
3 Mass Spectrometer	10 Water Tank	17 Extra Shoes and Clothing
4 Two Gas Bread-board Control	11 Nitrogen Tank	18 Over-garments
5 Lithium Hydroxide System	12 Oxygen Tank	19 Rescue Ball
6 Biomedical Station	13 IVA Backpacks	20 Sleeper
7 Water Pump	14 Food	21 Fires Suppression and Detection

Figure 6-3 (c)



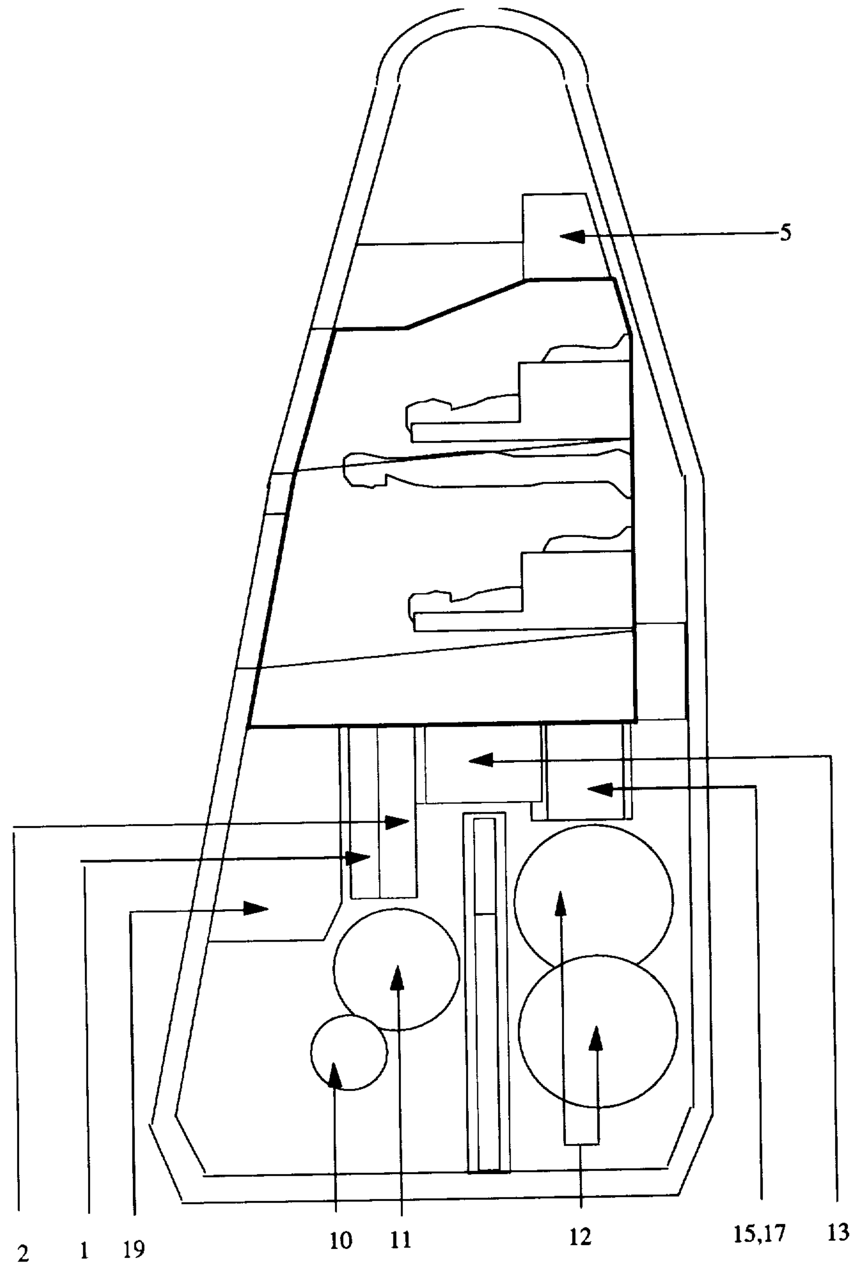
REAR PANEL VIEW 4 of 7	
1 Humidity Control	11 Nitrogen Tank
2 Thermal Control System	12 Oxygen Tank
3 Mass Spectrometer	13 IVA Backpacks
4 Two Gas Bread-board Control	14 Food
5 Lithium Hydroxide System	15 Toiletries
6 Biomedical Station	16 Tools and Cleaning Equipment
7 Water Pump	17 Extra Shoes and Clothing
8 Commode	18 Over-garments
9 Hygiene Station	19 Rescue Ball
10 Water Tank	20 Sleeper
	21 Fires Suppression and Detection

Figure 6-3 (d)



REAR VIEW	5 of 7
1 Humidity Control	12 Oxygen Tank
2 Thermal Control System	13 IVA Backpacks
3 Mass Spectrometer	14 Food
4 Two Gas Bread-board Control	15 Toiletries
5 Lithium Hydroxide System	16 Tools and Cleaning Equipment
6 Biomedical Station	17 Extra Shoes and Clothing
7 Water Pump	18 Over-garments
8 Commode	19 Rescue Ball
9 Hygiene Station	20 Sleeper
10 Water Tank	21 Fires Suppression and Detection
11 Nitrogen Tank	

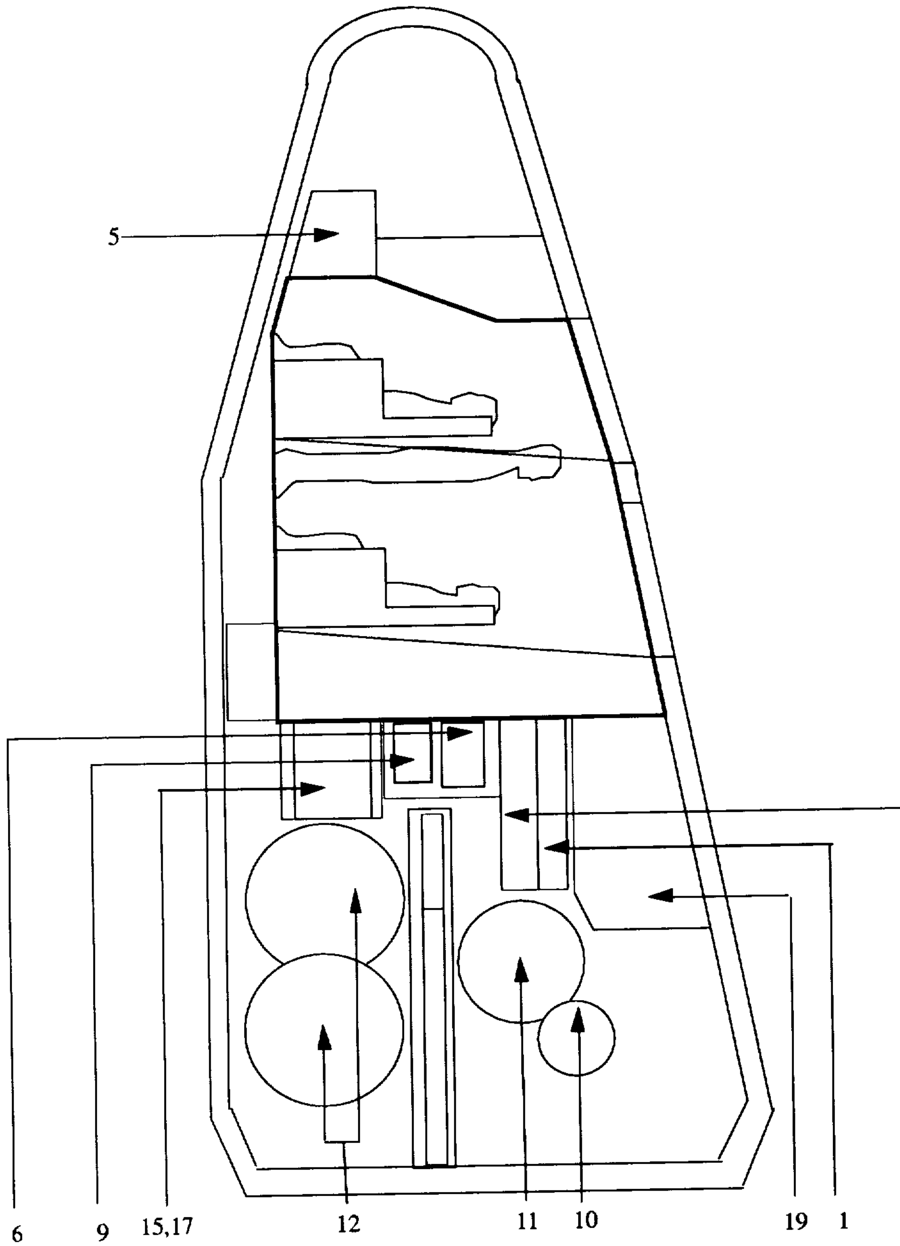
Figure 6-3 (e)



RIGHT SIDE VIEW 6 of 7

1 Humidity Control	8 Commode	15 Toiletries
2 Thermal Control System	9 Hygiene Station	16 Tools and Cleaning Equipment
3 Mass Spectrometer	10 Water Tank	17 Extra Shoes and Clothing
4 Two Gas Bread-board Control	11 Nitrogen Tank	18 Over-garments
5 Lithium Hydroxide System	12 Oxygen Tank	19 Rescue Ball
6 Biomedical Station	13 IVA Backpacks	20 Sleeper
7 Water Pump	14 Food	21 Fires Suppression and Detection

Figure 6-3 (f)



LEFT SIDE VIEW

7 of 7

1 Humidity Control	8 Commode	15 Toiletries
2 Thermal Control System	9 Hygiene Station	16 Tools and Cleaning Equipment
3 Mass Spectrometer	10 Water Tank	17 Extra Shoes and Clothing
4 Two Gas Bread-board Control	11 Nitrogen Tank	18 Over-garments
5 Lithium Hydroxide System	12 Oxygen Tank	19 Rescue Ball
6 Biomedical Station	13 IVA Backpacks	20 Sleeper
7 Water Pump	14 Food	21 Fires Suppression and Detection

Figure 6-3 (g)

Crew Systems Layout in CM

Placement of the oxygen, nitrogen, and water tanks along with other crew systems components provides a crew systems' center of gravity which helps achieve an overall center of gravity which satisfies aerodynamic structural restraints.

6.2.2.2 C3 Systems and GNC Systems Layout in the CM

As shown below, the components for GNC and C3 are located in the nose section of the CM, flush with the edge of the control panel:

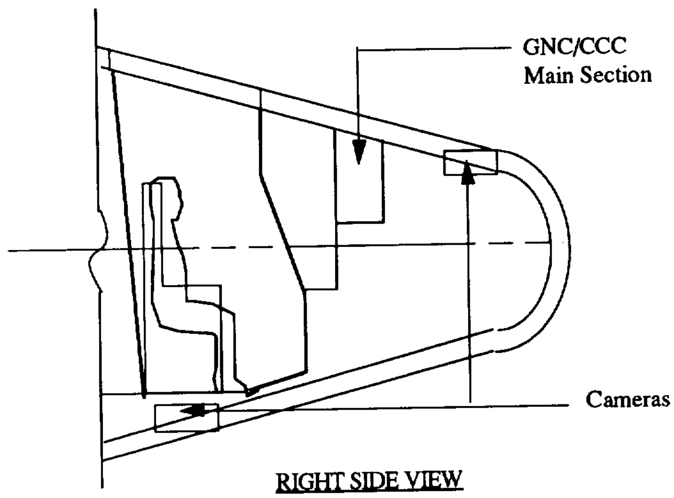


Figure 6-4
Location of Main GNC/CCC Systems

This allows the C3/GNC components to be easilty interfaced with the pilot control panel. These components are linked with the low-gain antennas and cameras mounted externally on the CM, and linked through the data bus, to the C3/GNC equipment onboard the ERM. Intra-systems interfaces are described in sections 6.4 and 6.5 of this chapter. A layout of these components which satisfies the intra-system interface requirements is shown in Figure 6-5.

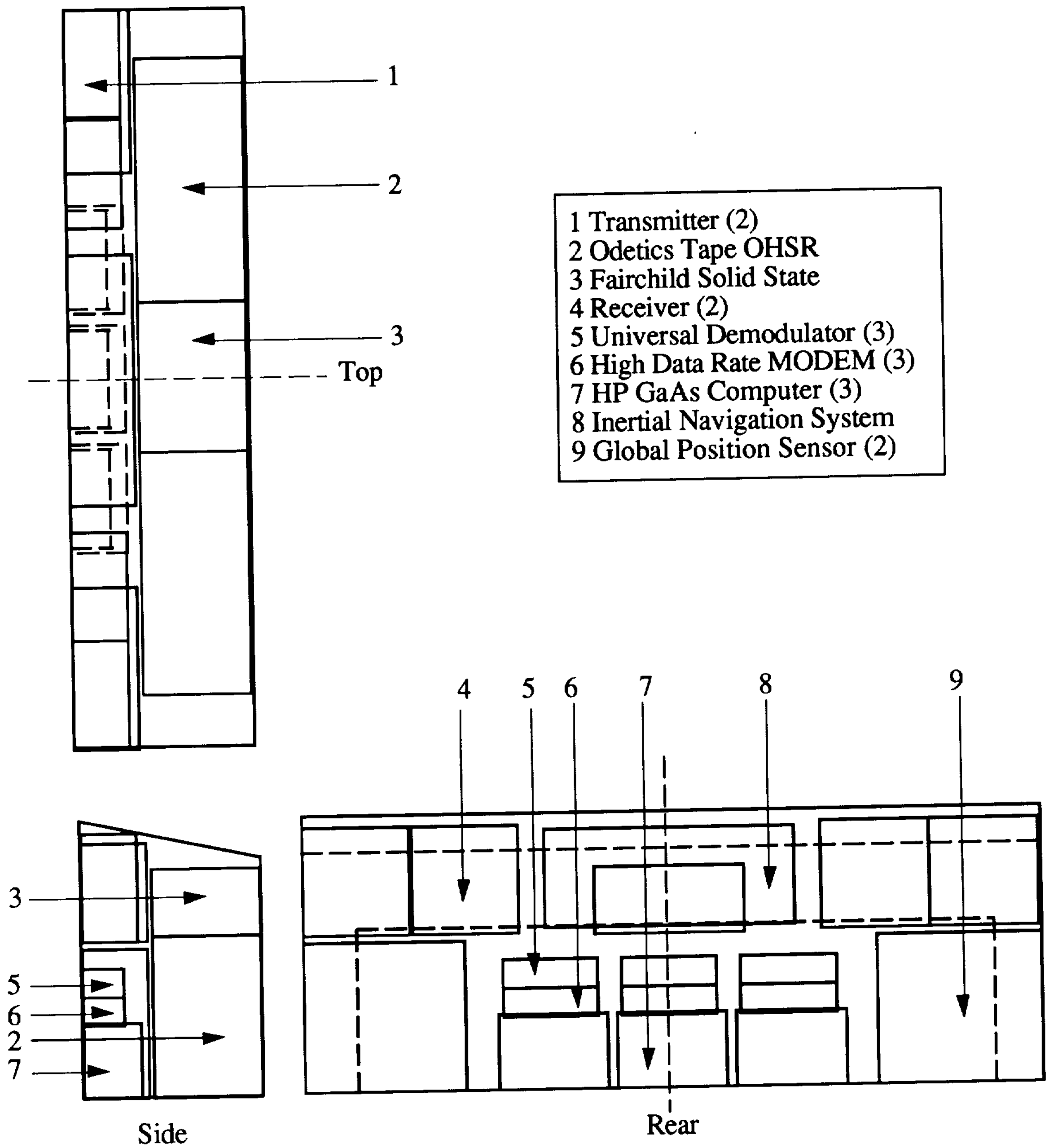
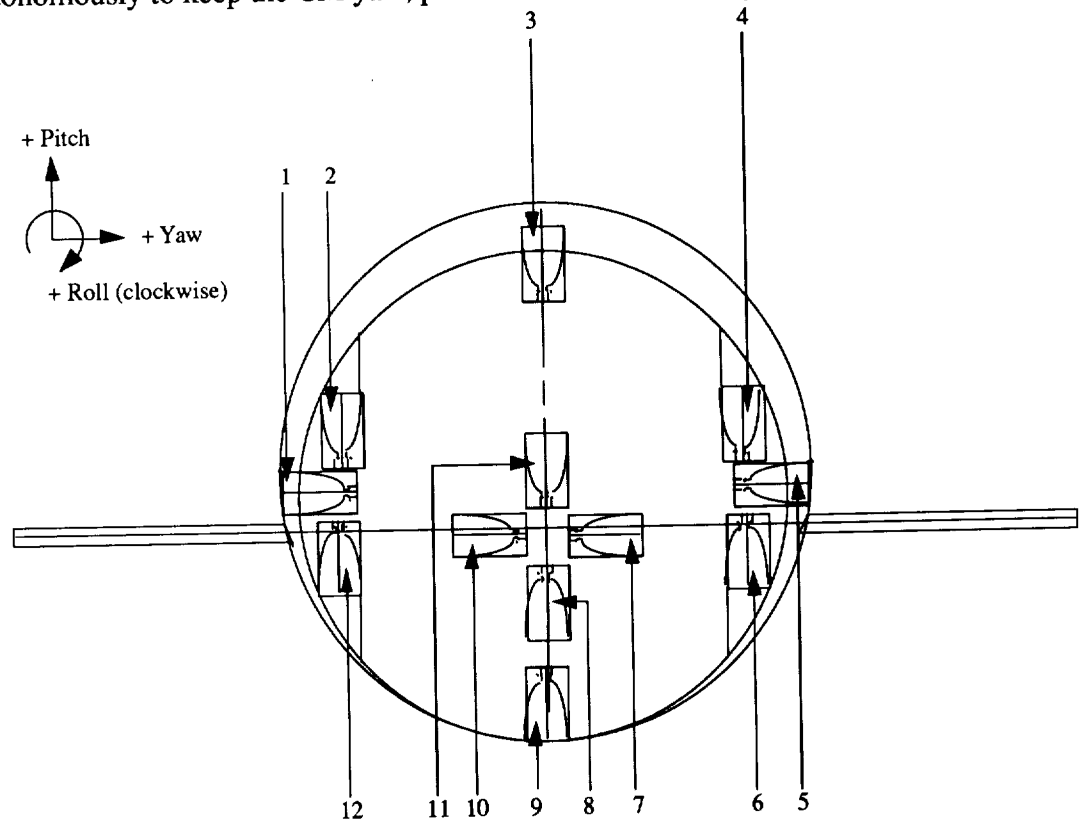


Figure 6-5
Layout for C3/GNC Components

Additional GNC equipment, including keyboards, liquid crystal displays, cathode ray tubes, and joysticks are located on the control panel.

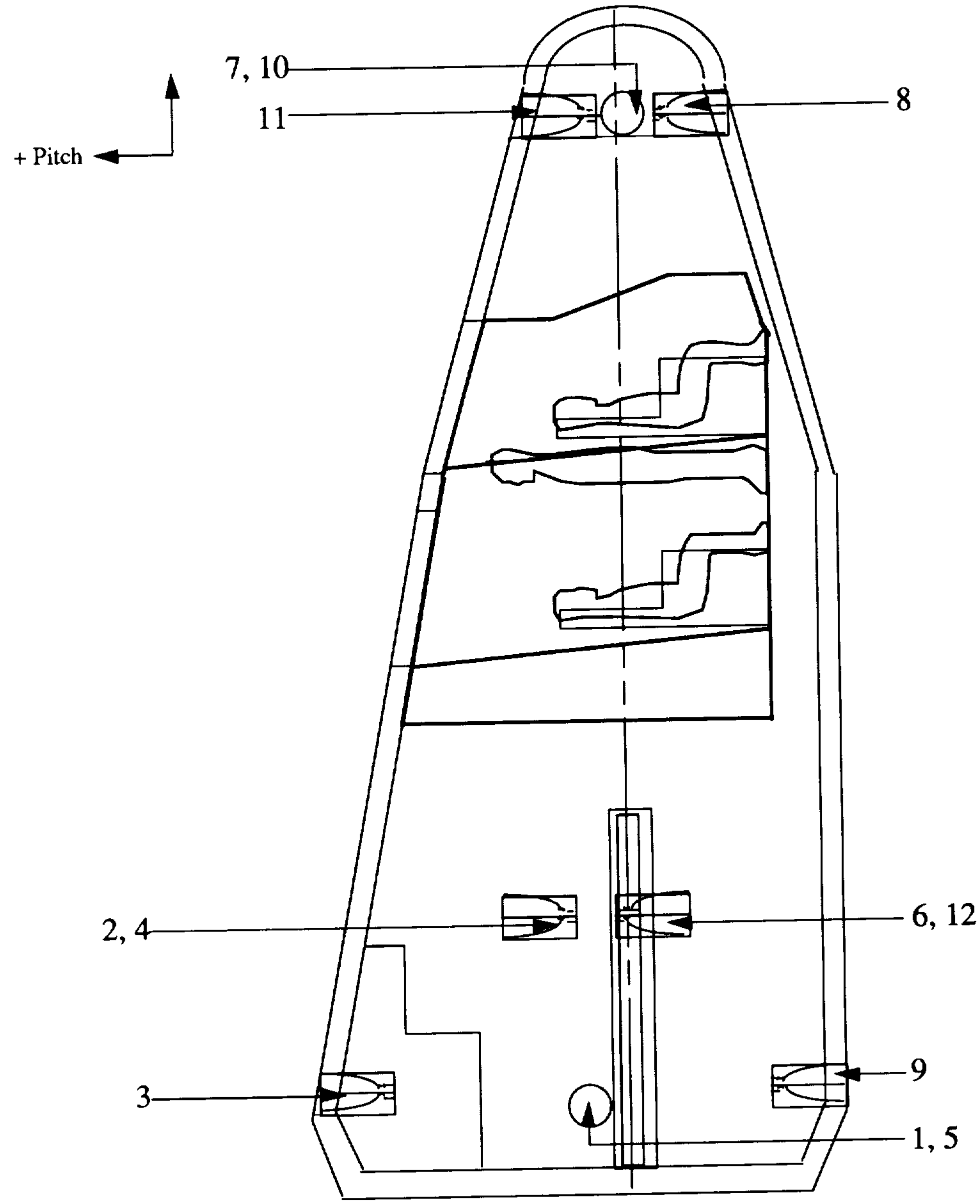
6.2.2.3 Reaction Control Systems Layout in the CM

Figure 6-6 (a) through (f) shows the component placement of the RD-4 RCS engines. The CM design requires 8 of the 12 RCS engines shown, and 400kg of fuel is prvided for corrections during the descent. The RCS system displayed provides for positive and negative pitch, right and left yaw, and right and left roll corrections in the CM trajectory during its reentry into the Earth's atmosphere. The pitch correction engines are shown but are not part of the needed design. The biconic shape has been determined to be pitch stable during reentry. The RCS engines are linked to the data bus network on the CM and operate autonomously to keep the CM yaw, pitch and roll stable during the descent.



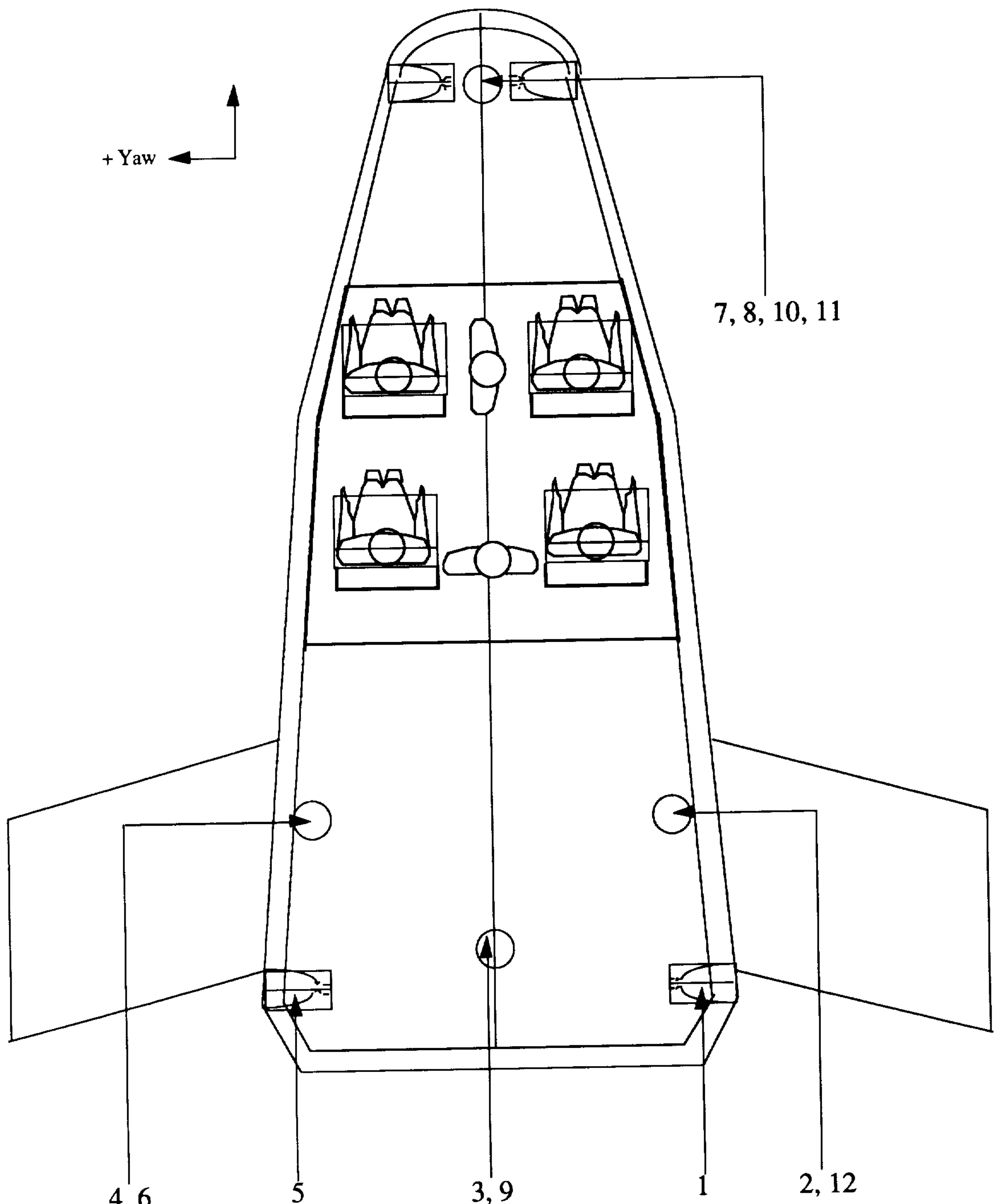
FRONT VIEW			1 of 6
1	Negative Yaw RCS Engine	7	Negative Yaw RCS Engine
2	Counterclockwise Roll RCS Engine	8	Positive Pitch RCS Engine
3	Positive Pitch RCS Engine	9	Negative Pitch RCS Engine
4	Clockwise Roll RCS Engine	10	Positive Yaw RCS Engine
5	Positive Yaw RCS Engine	11	Negative Pitch RCS Engine
6	Counterclockwise Roll RCS Engine	12	Clockwise Roll RCS Engine

Figure 6-6 (a)



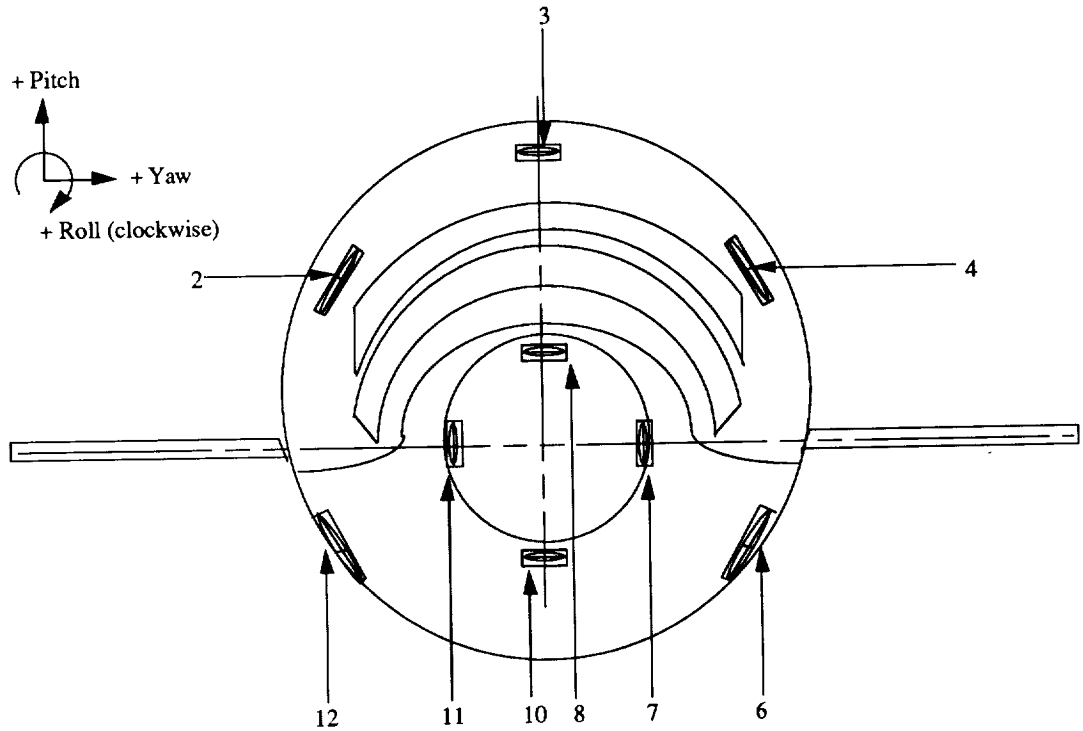
RIGHT SIDE VIEW			2 of 6
1	Negative Yaw RCS Engine	7	Negative Yaw RCS Engine
2	Counterclockwise Roll RCS Engine	8	Positive Pitch RCS Engine
3	Positive Pitch RCS Engine	9	Negative Pitch RCS Engine
4	Clockwise Roll RCS Engine	10	Positive Yaw RCS Engine
5	Positive Yaw RCS Engine	11	Negative Pitch RCS Engine
6	Counterclockwise Roll RCS Engine	12	Clockwise Roll RCS Engine

Figure 6-6 (b)



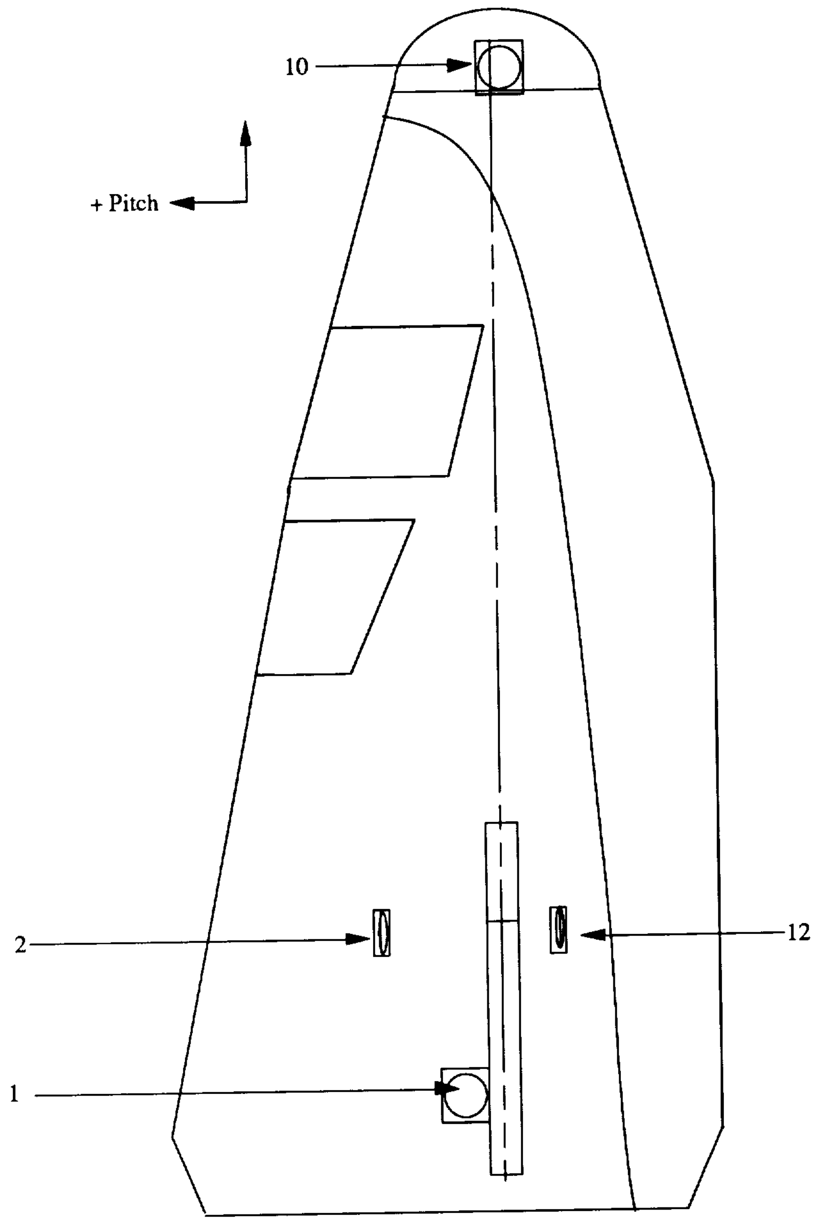
TOP VIEW			3 of 6
1	Negative Yaw RCS Engine	7	Negative Yaw RCS Engine
2	Counterclockwise Roll RCS Engine	8	Positive Pitch RCS Engine
3	Positive Pitch RCS Engine	9	Negative Pitch RCS Engine
4	Clockwise Roll RCS Engine	10	Positive Yaw RCS Engine
5	Positive Yaw RCS Engine	11	Negative Pitch RCS Engine
6	Counterclockwise Roll RCS Engine	12	Clockwise Roll RCS Engine

Figure 6-6 (c)



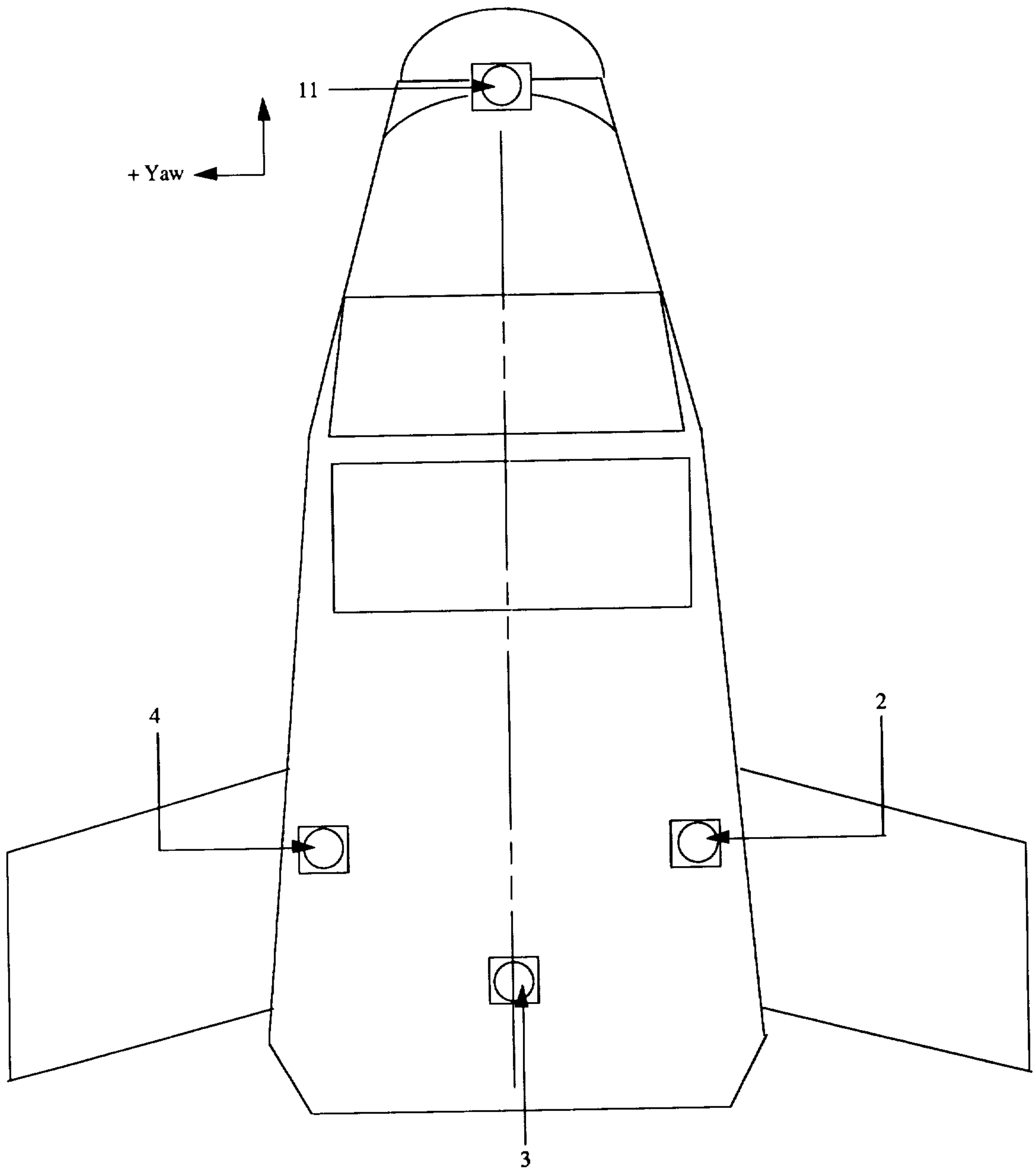
FRONT EXTERIOR VIEW			5 of 6
1	Negative Yaw RCS Engine	7	Negative Yaw RCS Engine
2	Counterclockwise Roll RCS Engine	8	Positive Pitch RCS Engine
3	Positive Pitch RCS Engine	9	Negative Pitch RCS Engine
4	Clockwise Roll RCS Engine	10	Positive Yaw RCS Engine
5	Positive Yaw RCS Engine	11	Negative Pitch RCS Engine
6	Counterclockwise Roll RCS Engine	12	Clockwise Roll RCS Engine

Figure 6-6 (d)



RIGHT SIDE EXTERIOR VIEW			5 of 6
1	Negative Yaw RCS Engine	7	Negative Yaw RCS Engine
2	Counterclockwise Roll RCS Engine	8	Positive Pitch RCS Engine
3	Positive Pitch RCS Engine	9	Negative Pitch RCS Engine
4	Clockwise Roll RCS Engine	10	Positive Yaw RCS Engine
5	Positive Yaw RCS Engine	11	Negative Pitch RCS Engine
6	Counterclockwise Roll RCS Engine	12	Clockwise Roll RCS Engine

Figure 6-6 (e)



TOP EXTERIOR VIEW			6 of 6
1	Negative Yaw RCS Engine	7	Negative Yaw RCS Engine
2	Counterclockwise Roll RCS Engine	8	Positive Pitch RCS Engine
3	Positive Pitch RCS Engine	9	Negative Pitch RCS Engine
4	Clockwise Roll RCS Engine	10	Positive Yaw RCS Engine
5	Positive Yaw RCS Engine	11	Negative Pitch RCS Engine
6	Counterclockwise Roll RCS Engine	12	Clockwise Roll RCS Engine

Figure 6-6 (f)

RCS Engine Locations on the Crew Module

6.2.3 Interfaces with the Crew Module

The modules that the CM interfaces with are the ERM (Earth Return Module) and the Biocan.

6.2.3.1 ERM Interface with the Crew Module

There are three ways in which the CM interfaces with the ERM: 1) through the power and support systems, 2) through the data bus systems for sensing, controlling, transmitting, and recieving to the Earth and the other modules, and 3) through the physical connection between the two modules which exists until right before the CM prepares to reenter into the Earth's atmosphere. Figure 6-7 shows the CM/ERM interface.

6.2.3.1.1 Crew Support and Power Systems

The CM and the ERM remain attached for the entirety of the lunar mission. They separate right before the CM reenters to land while the ERM burns up on its own reentry. Since they remained attached for all this time, it is beneficial to divert some of the CM's support systems to the ERM without complicating either of the designs. The current design of the ERM includes additional space for 2000 kg of payload. This will be used for delivering additional crew support systems (mostly consumables) needed during the stay on the Moon. Designing the ERM to store these consumables needed while on the Moon for the first mission allows for its design to be used to deliver those consumables needed for the next return mission to the lunar habitat. This approach is easier and costs less than leaving these systems on the Biocan launch and having to design an extra module to deliver this needed payload for the next mission, thereby improving the expandability of the project overall.

These specific support systems can be stored in the ERM: water, oxygen, nitrogen, food, clothing, tools and scientific experiments and equipment.

The current design has most of the power systems for the CM located in the ERM. Removing these power systems from the CM make it less self sustaining, but removing these systems also reduces the overall reentry weight.

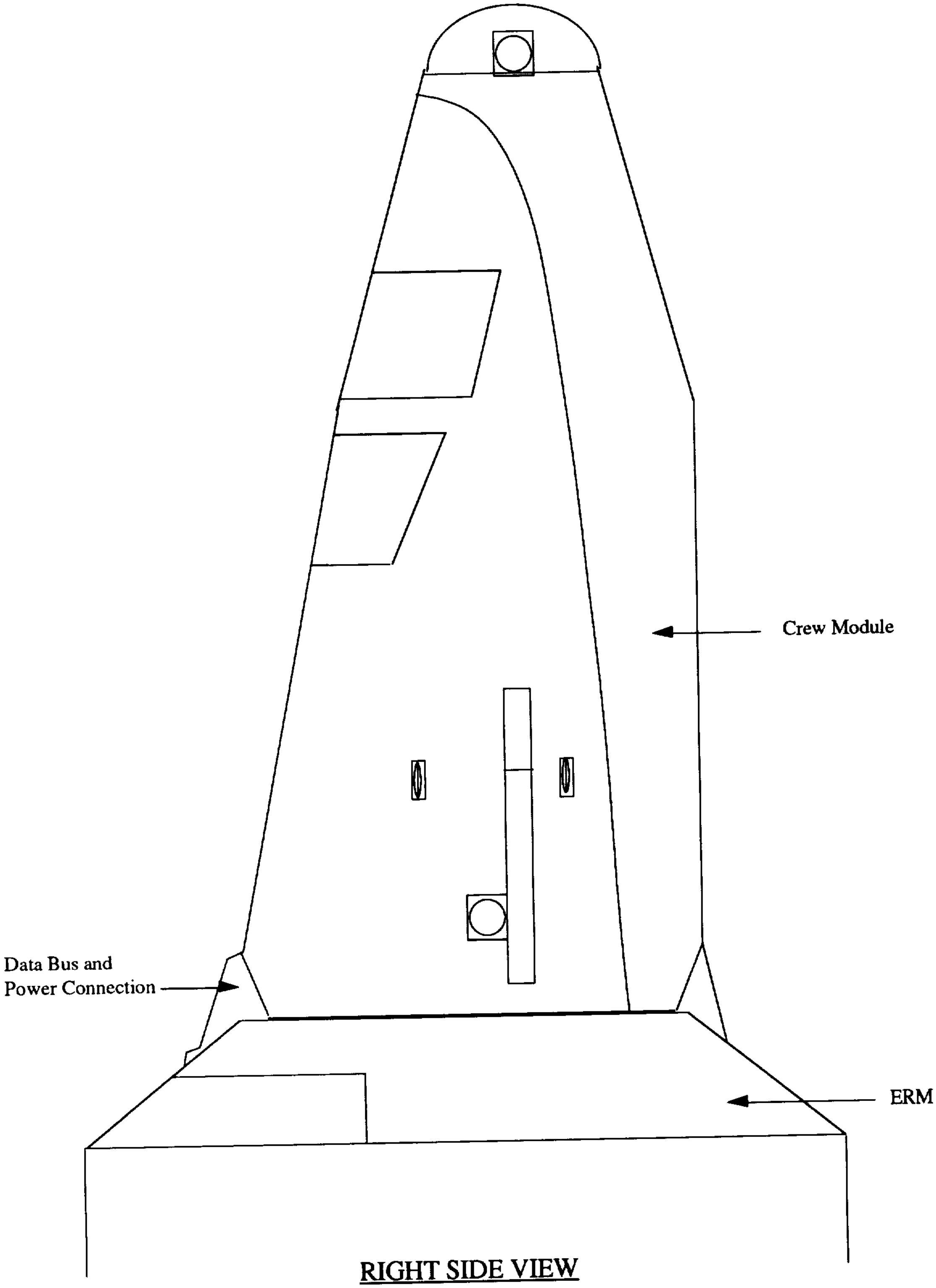


Figure 6-7
Diagram of the CM/ERM Interface

6.2.3.1.2 Data Bus Connection

Along with the crew support and power system interfaces that can exist between the CM and the ERM, there exist the necessary sensing and controlling connections to the systems from the TLI (Trans-lunar Injection) stage, the LBM (Lunar Braking Module) stage, and the ERM stage to the CM. These connections allow the crew to conduct a self-check on all of the systems on the different modules and correct any problems that they can.

All sensors, actuators, engines and other components are linked to the main data bus. The main flight control panel is also linked to the data bus, and allows for the crew to perform a system override or mission abort if neeeded.

6.2.3.2 Habitat Interface with the Crew Module

Although the crew will live in a habitat throughout the 28 day lunar stay, the CM must be kept idle and maintainanced often in order to assure that the lunar stay may be aborted at any time.

6.2.3.2.1 Servicing the Crew Module

The habitat could have been designed to store support systems that the for CM that the crew would use for the return mission. These support systems (such as consumables for crew and power for the instruments) could have been secured in the habitat and brought over into the CM when needed. The advantage of having the CM depend on the Biocan for support sytems for the transit back to Earth is that launched weight of the CM is less, making it easier to meet the 6000 kg payload to lunar surface requirement, but the disadvantage is that in the case of an abort via to the lunar surface, these needed support systems for the crew to return to earth will never be obtained from the habitat. That would endanger the feasibility of an important abort mode, so the interfacing of power and consumables for the return flight between the CM and habitat will not be implemented in the design of this lunar mission.

The only possible things that can be stored on the Biocan without endangering the mission are those systems that the crew will specifically use while on the lunar surface that cannot be readily stored in the ERM, such as the crew members' hardsuits for the stay on the surface. If these systems become unattainable due to a failure of operation of the habitat, then the crew still has the ability to make it back to the earth safely.

6.2.4 Centriod Calculations for the CM

As mentioned, one of the key drivers in designing the CM layout was obtaining a mass centroid which would satisfy the aerodynamic requirements for a successful landing. Throughout the entire stage of designing the layout of the each subsystem and their respective components, the mass centroid of the CM and the individual subsystems was continuously updated. If a current layout did not meet the centroid requirements, as well as all other layout design requirements, it was eliminated and the design iteration process continued.

6.2.4.1 Mass Centroid Requirements

The analysis of lift coefficients of the biconic CM revealed that the mass centroid should lie along the longest axis of the spacecraft, approximately 1/2 of a chord length past the leading edge of the deployable wings. For the CM shell of the selected geometry this would place the mass centroid at approximately 3 meters away from the back of the spacecraft. It was enforced that this centroid requirement was a minimum value needed in order for the wings to function as designed; although it was also stressed that the centroid could not be too far from the wings' leading edges or else the spacecraft would descend at too steep of an angle. Therefore, a centroid design goal (along the long axis of the CM, or the x direction) of 3.0 to 3.5 meters away from the back of the CM was established .

Original centroid goals of the other two axes of the CM were 0 m away from the centerline of the spacecraft. However, the roll stability of the CM during initial reentry, before the wings were deployed, was a concern. If dynamic instability caused the CM to roll over, exposing its unprotected top to the searing heat of reentry, the craft would "burn up." To account for this possibility, an attempt was made to design the overall layout so that the centroid along the bottom to top center line axis (the z-direction) of the CM would be slightly negative. In this case, the CM would act as a pendulum: it the craft rolled a few degrees a moment would be created in the opposite direction, returning the CM to its original orientation. It was decided that the centroid in the z-axis could not be positive because the CM would act as an inverted pendulum, naturally inclined to turn over after a small displacement. It was also decided that the CM centroid along the starboard to port axis (the y-direction) should also be zero, so not as to induce any natural inclinations to freely roll, before the wings were deployed.

6.2.4.2 Method of Mass Centoid Determination

The first step in determining the mass centroid was to define an orthogonal set of axes based at an origin point on the body. The axis directions were defined as:

- •X-Direction: along the longest center-line of the CM, with positive direction running from aft to fore.
- •Y-Direction: along the starboard to port center-line of the CM, with positive direction running for starboard to port.
- •Z-Direction: along the bottom to top center-line of the CM, with positive direction runnnig upward.

The origin of the CM was defined as the point in the center of the maximum diameter (where the CM interfaces with the ERM) at the intersection point of the diameter. The origin point was set on the back inner wall of the CM.

The second step was representing all of the subsystems and their components as point bodies. This was done by representing each component as a rectangle, cube, sphere, or prism of approximate shape and dimensions or approximate volume, whichever was available. Each or these shapes were fitted into the inner volume of the CM, based on where each part was supposed to be located in order to interface with other parts and the crew members, as well as satisfying the volume and centroid limitations. In section 6.2.2 of this chapter, diagrams showing the placement of components for Crew Systems; Guidance, Navigation, and Control; and Command, Control, and Communications were displayed. Figure 6-8 shows an exact diagram, depicting the location of all components within the CM (with the exception of the control panel and the main CCC/GNC bay). These compnents were exactly fitted into the inner volume of the CM using the Claris computer-aided-drafting program. Claris CAD™ is able to give the center coordinates of each depicted shape of exact dimensions. These coordinates were then used to represent the point mass representation of each component. This same procedure was used for enlarged drawings of the main CCC/GNC bay and the control panel.

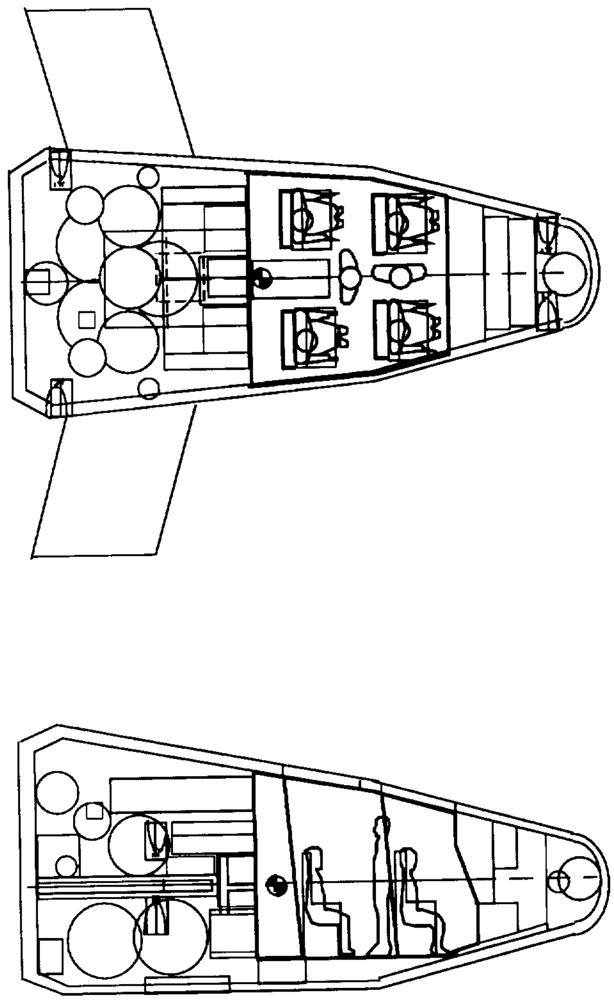


Figure 6-8 (a)
Right Side and Top Cut-away Views of CM

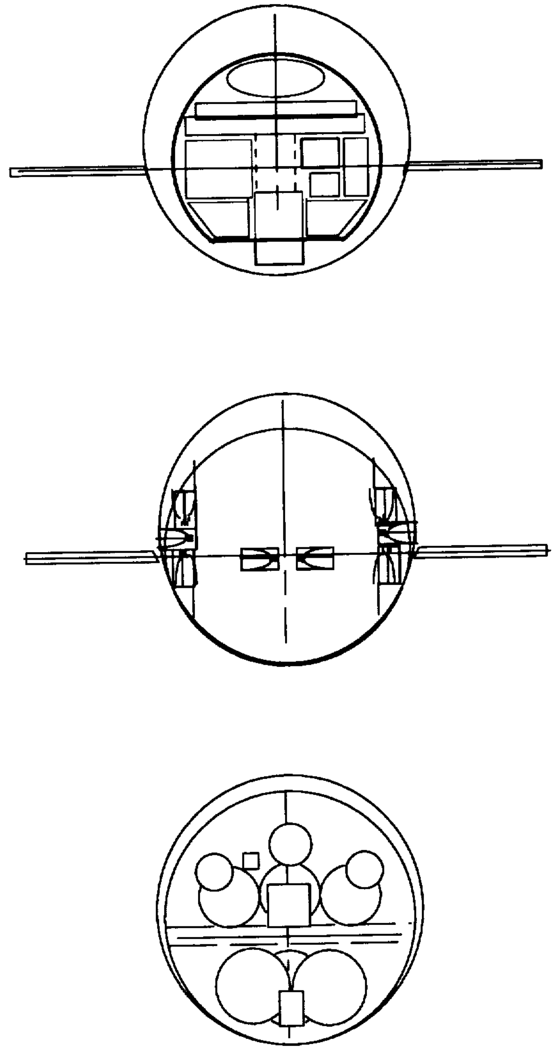


Figure 6-8 (b)

Panel Views of the Components within the CM

Note: Claris CAD representation of components within the CM. X-axis lies along longest center-line of the CM; Y-axis along the starboard to port center-line; Z-axis from the bottom to top center line. Orign lies at center of maximum diameter on inner wall. All dimension and object centers recorded by Claris CAD. Centroid is marked on Top and Right Side views. For control panel and CCC/GNC components, please see figures 6-2 and 6-4. Coordinates of centroid location is (x,y,z) = (3.14, 0.00, -0.02) meters.

The above representations are used to demonstrate that all components fit within the volume constraint.

6.2.4.3 Mass Centoid Results

In order to represent the centroids of the capsule, the CM was first divided up by sub-system. Below are tables representing the centroid calculations of the Crew Systems, GNC/CCC, Control Panel, and RCS subsystem components.

Table 6-2 (a): Crew Systems Centroid of Mass

Component/Sub-system	Mass (kg)	Vol(m^3)	Position (m) X	Y	Z
Main Cabin					
Sleeper 1	16	0.6	4.55	0.7	-0.24
Sleeper 2	16	0.6	4.55	-0.7	-0.24
Fire Suppression and Detection	25	0.05	4.3	0	0
Lighting	4	0.01	4.3	0	0
Aft Top Section (A)					
Humidity Control	55	0.255	2.378	-0.5	0.745
Thermal Control	70	0.65	2.375	0	0.53
Water 1	60.17	0.06	0.76	0.8	0.87
Water 2	60.17	0.06	0.76	-0.8	0.87
Mass Spectrometer	18.2	0.25	2.378	0.5	0.745
2-Gas Breadboard Control	22.7	0.2	2.67	0	0.04
Biomed Equipment	22.98	0.08	2.73	0.56	0.18
Nitrogen 1	42.25	0.27	1.35	0	0.54
Nitrogen 2	42.25	0.27	1.35	0.82	0.54
Nitrogen 3	42.25	0.27	1.35	-0.82	0.54
water pump	10	0.008	1.24	-0.5	1.04
Aft Bottom Section (B)					
Commode	46	0.24	2.625	0	-0.66
Oxygen 1	35.3067	0.53	0.91	0.5	-0.72
Oxygen 2	35.3067	0.53	0.91	-0.5	-0.72
Oxygen 3	35.3067	0.53	1.77	0	-0.72
Hygiene Station	20	0.03	2.73	0.585	-0.24

Storage					
backpacks	32	0.3	2.67	-0.82	-0.01
Food	22	0.2	3.255	0	-1.15
Toiletries	3	0.03	2.36	-0.68	-0.66
Tools, cleaning equipment	29.5	0.11	2.67	1	-0.04
shoes & clothing	22.4	0.2	2.63	-0.7	-0.66
Over-garments	45.4	0.13	2.36	0.68	-0.66
Rescue-ball	2	0.128	2.78	0	1.22
Fore Section (C)					
LiOH (Main)	20	0.2	6.165	-0.3	-0.57
TOTALS	855.19	6.79			

Cx is	Cy is	Cz is
2.16	0.01	0.09

Table 6-2 (b) : GNC/CCC Main Bay Centroid of Mass

Component/Sub-system	Mass (kg)	Volume (m^3)	Position (m) X	Y	Z
HP GaAs Computer 1	25	0.00303	6.1735	0	0.15
HP GaAs Computer 2	25	0.00303	6.1735	-0.22	0.15
HP GaAs Computer 3	25	0.00303	6.1735	0.22	0.15
Odetics Tape OHSR	45.4	0.0708	6.35	0	0.23
Fairchild Solid State	6.17	0.00684	6.35	0	0.441
Universal Demodulator 1	20.4	0.00068826	6.1581	0	0.2965
Universal Demodulator 2	20.4	0.00068826	6.1581	-0.22	0.2965
Universal Demodulator 3	20.4	0.00068826	6.1581	0.22	0.2965
High Data Rate Modem 1	10	0.00068826	6.1581	0	0.2954
High Data Rate Modem 2	10	0.00068826	6.1581	-0.22	0.2954
High Data Rate Modem 3	10	0.00068826	6.1581	0.22	0.2954
Transmitter 1	1	0.004	6.17	-0.58	0.47
Transmitter 2	1	0.004	6.17	0.58	0.47
Receiver 1	1	0.004	6.17	-0.38	0.47
Receiver 2	1	0.004	6.17	0.38	0.47
INS	10	0.01	6.18	0	0.46
GPS 1	10	0.01	6.18	0.53	0.22
GPS 2	10	0.01	6.18	-0.53	0.22
TOTALS	251.77	0.14			

Cx is:	Cy is:	Cz is:
6.20 m	0.00 m	0.25 m

Table 6-2 (c) : RCS Mass Centroid

Component/Sub-system	Mass (kg)	Volume (m^3)	Position (m) X	Y	Z
RD-4 #1	3.63	0.0475	7.025	0.5	-0.05
RD-4 #2	3.63	0.0475	7.025	-0.05	-0.05
RD-4 #3	3.63	0.0475	0.45	1.36	0.15
RD-4 #4	3.63	0.0475	0.45	-1.36	0.15
RD-4 #5	3.63	0.0475	1.52	1.3	0.55
RD-4 #6	3.63	0.0475	1.52	-1.3	-0.55
RD-4 #7	3.63	0.0475	1.52	1.3	0.55
RD-4 #8	3.63	0.0475	1.52	-1.3	-0.55
TOTALS	29.04	0.33			

Cx is:	Cy is:	Cz is:
2.63 m	0.00 m	0.03 m

Table 6-2 (d) : Control Panel Mass Centroid

Instrument	Mass (kg)	Volume (m^3)	Position (m) X	Y	Z
Joystick 1	1	0.003	5.07	0.74	-0.68
Joystick 2	1	0.003	5.07	-0.74	-0.68
CRT1	10	0.0553	6.285	0.62	0.3
CRT2	10	0.0553	6.285	-0.62	0.3
LCD1	3	0.00277	5.95	1.3	0.3
LCD2	3	0.00277	5.95	-1.3	0.3
Keyboard 1	3	0.00629	5.849	1.3	-0.1
Keyboard 2	3	0.00629	5.849	-1.3	-0.1
Camera 1	4	0.002	4.9	0.4	-0.218
Camera 2	4	0.002	4.9	-0.4	-0.218
Camera 3	4	0.002	8.7	0.2	1.05
Camera 4	4	0.002	8.7	-0.2	1.05
TOTALS	50	0.14			

Cx is:	Cy is:	Cz is:
6.31	0	0.25

Table 6-2 (e) : Overal CM Centroid of Mass

Component/Sub-system	Mass (kg)	Volume (m^3)	Position (m)		
			X	Y	Z
Mission Commander	75	2.5	4.9	0.74	-0.18
Pilot	75	2.5	4.9	-0.74	-0.18
Mission Specialist 1	75	2.5	3.62	0.74	-0.18
Mission Specialist 2	75	2.5	3.62	-0.74	-0.18
Ejection Seat 1	94.51	0.53	4.9	0.74	-0.68
Ejection Seat 2	94.51	0.53	4.9	0.74	-0.68
Ejection Seat 3	94.51	0.53	3.62	-0.74	-0.68
Ejection Seat 4	94.51	0.53	3.62	-0.74	-0.68
Crew Systems	855.19	6.79	2.16	0.01	0.09
CCC/GNC Main Section	251.77	0.14	6.2	0	0.25
Instrument Panel	50	0.14	6.31	0	0.25
Structural Shell	1800	n/a	2.9	0	0.23
Heat Shield	732	n/a	3.24	0	-0.64
Wing (Starboard)	160	0.7	1.1	-0.54	0
Wing (port)	160	0.7	1.1	0.54	0
Landing Gear 1	45	0.11	4.62	0.3	-1.08
Landing Gear 2	55	0.11	1.52	-0.3	-1.12
Landing Gear 3	55	0.11	1.52	0	-1.12
Drogue Chutes	20.41187	0.19	0.26	0	0.66
Parafoil	45.8	0.57	6.2	0	0.4
O/F/Pressurization Aft	200	??	0.4	0	1.4
O/F/Pressurization Fore	200	??	7.24	0	0
RCS Engines	29.09	0.33	2.63	0	0.03
Power Systems	21.6	0.0414	0.15	0	-1.41
TOTALS	5358.902	22.05			

Cx is:	Cy is:	Cz is:
3.14	0	-0.02

The resulting mass centroid of the CM total layout, (Cx,Cy,Cz) = (3.14 m, 0 m, -0.02 m), satisfies all mass properties requirements.

6.2.5 Abort System Design and Implementation

An ejection seat system was chosen as the method of launch escape for the Project Columbiad crew capsule. Ejection seats were chosen over a launch escape tower system because of easier capsule integration in the biconic design, lower cost, lower weight, proven reliability, reduced developmental risk, reusability, maintainability, and post-reentry escape availability.

The CM will require the implementation of four Mk-14 NACES ejection seats. These ejection seat units include the parachute, guide rails for a stable exiting of the CM, deployable pitot tubes to determine the ejection parameters, a survival kit located under the seat, a flotation device, drogue chutes for stability after the seat leaves the guide rails, and restraints for the crew members' legs and arms to avoid injury. The unit is autonomously controlled after ejection, has some crew support systems such as oxygen, flares, and high visibility communication systems. The seats will be attached to the base of the CM, and seats and rails will be oriented such that the units do not run the risk of colliding after ejection. The crew members will wear IVA (Inter Vehicular Activity) suits while they are strapped in to their seats. These suits can withstand depressurization as well as provide anti-G protection for the crew member. In the case of an abort before the launch vehicle reaches an altitude of 36 km, all four ejection seat units can be activated from both the Mission Commander's seat and Ground Control.

The Mk-14 ejection seat units have many advantages. The ejection seat units weight more than a regular g-couch unit would, but they provide an abort mode option for reentry as well as launch. This extra option greatly improves the chances of the crew's survival for the mission. The cost of development integrating the Mk-14 ejection seats into the CM is significantly less than the amount of funding needed to develop a launch escape tower, and the units are fairly self-sustaining and are a proven successful technology.

The disadvantage in using the ejection seats is that there is a brief window in the mission profile while the SRB's (Solid Rocket Boosters) are burning and there is no scheduled abort mode - the CM has passed the 36 km altitude limit on the operation of the ejection seats and the ERM module cannot provide enough thrust to push away from the SRB stack until the boosters stop burning. Designing to make this window shorter will improve the survivability of the crew during this period.

6.2.5.1. Crew Ejection Seat Subsystem Description

The ejection system chosen for Project Columbiad is the Martin-Baker Mk-14 Navy Aircrew Common Ejection Seat (NACES) Update II version (see Figure 3-10). This state-of-the-art escape system is currently installed in the F/A-18 Hornet, F-14D Tomcat, and T-45 Goshawk airframes. A major feature of this seat is the incorporation of a microprocessor-based electronic sequencing system. The seat is qualified for operation between 0 and 308 m/s equivalent airspeed (0 to 600 KEAS) and all altitudes under 36 km (120,000 ft) with the incorporation of a full-pressure suit as standard flight equipment. A single Mk-14 seat weighs 87 kg and displaces a volume of 135 x 51 x 77 cm (0.53 m^3). The Mk-14 Update I seat is currently in full production, and the Update II version is expected to enter production within the next 2-3 years; available estimates indicate that a complete escape system can be purchased and installed for a cost on the order of $500,000.

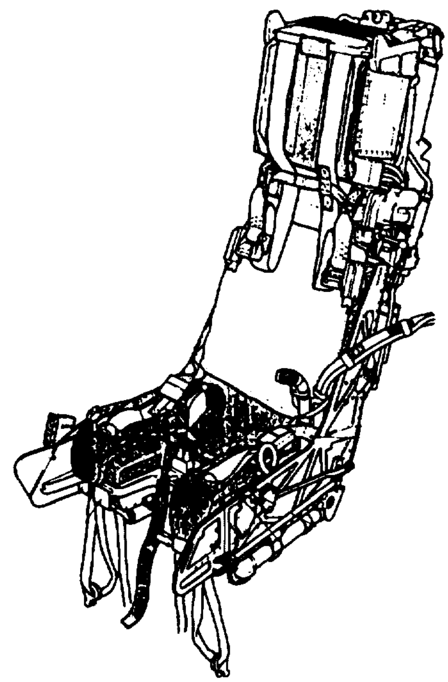


Figure 6-9
Mk-14 NACES Update II Ejection Seat

The Mk-14 NACES ejection seat provides improved performance over other ejection seat designs by incorporating several new and innovative design features. A quick-acting ribbon drogue is deployed as the seat leaves the guide rails to stabilize the seat in the pitch and yaw axes. Deployable pitot heads are ballistically deployed outboard following spacecraft egress. These devices sense the ejection conditions to enable the sequencer to determine the ejection parameters. A microprocessor controlled electronic sequencer matches pitot information with the preprogrammed tables and selects the appropriate mode of operation to suit the ejection conditions. Additionally, the seat electronic sequencer operation is enhanced by interconnection with the spacecraft databus, allowing electronic decision-making and mode selection to begin prior to pitot deployment. All post-ejection functions are electrically commanded by the sequencer which is powered by redundant long-lasting thermal batteries. The main-canopy deployment drogue and personnel parachute are deployed by a small tractor-rocket extraction system which deploys the canopy in a reefed condition prior to main opening. This system in particular (i.e., the chute deployment system) offers an improved pad ejection capability for the Project Columbiad astronauts.

Passive arm and leg restraints are incorporated into the Mk-14 Update II seat to prevent limb flailing and injuries at high speed. The crewmember services (anti-G, oxygen, air ventilation and communications) are routed via a single action disconnect block for seat/crewmember separation upon ejection. The single-point parachute harness and parachute are integral with the seat, obviating the need for a separate harness to be carried on the astronaut's pressure suit. Main canopy disconnect, survival kit deployment, and liferaft inflation are automatically initiated using the Seawater Activated Release System (SEAWARS) should the astronaut experience an ocean landing. A weight/item breakdown for individually-carried and seat-mounted survival kit items is included in Table 6-3. Given these preliminary estimates, the following survival equipment budgets are recommended: 1.5 kg/person for pressure suit-mounted survival equipment; 2.2 kg/person for liferaft-related gear; 4.0 kg/person for seat-mounted survival equipment. Pressure suit-mounted survival equipment is designed to ensure a minimum of 6 hrs. survival time, while the liferaft and seat-mounted gear extends survival time in excess of 24 hours. Previous NASA studies have indicated that 24 hours is the maximum response time which Air Force/Navy Search and Rescue forces feel is necessary to locate downed crewmembers in the event of a contingency abort. In addition, all survival equipment meets current NASA requirements for emergency spacecraft ditching along projected launch ground tracks (worst case environment: water temp. = 4.4°C, air temp. = 5.6°C, 1 foot waves (chop), and constant spray).

The Mk-14 ejection seat is modular in construction and easy to maintain. Depot-level maintenance for the NACES system is required only on a 3-year + basis.

The ejection sequence is initiated following a determination that a hazardous situation exists. Initiation is either commanded via electronic signal from the Range Safety Officer or by manual activation by the Mission Commander or other crewmember (a decision by the Mission Commander to eject will eject all other crewmembers automatically). The seat-mounted center pull handle fires redundant ejection initiator cartridges which activate the mechanical and electrical ejection mechanisms. Further ejection operations, including rocket motor firing, attitude adjustment, drogue deployment, seat/crewmember separation, main canopy deployment, and recovery are automatic.

Table 6-3: Survival Kit Equipment Budget

Pressure Suit Mounted

Strobe Light - SDU-5/E	0.21 kg
Pen Gun Flare Set	0.20 kg
Day/Night Flare - Mk-13/Mk-124	0.19 kg
Signal Mirror	0.05 kg
Drinking Water - (2) 4 oz.	0.34 kg
Leatherman Tool (modified)	0.14 kg
Medical Kit (bandage, anti-bacterial agent, sunblock)	0.20 kg
Whistle	0.03 kg
Chemlites (2) - 1 ultrahigh intensity	0.10 kg

	1.46 kg

Seat Mounted

Navy LRU-18/U Liferaft with NASA Spray Shield	2.10 kg

	2.10 kg

Seat Mounted Survival Kit

Emergency Locater Transponder - URT-33	0.34 kg
Day/Night Flare - Mk-13/Mk-124	0.19 kg
Radar Reflector - Balloon Type	0.08 kg
Signal Panel/Survival Blanket	0.11 kg
Dye Marker (2)	0.20 kg
Chemlites (2) - 12 hr type	0.10 kg
Drinking Water - (5) 4 oz.	0.85 kg
Reflective Strip - (2) for Raft	0.05 kg
First Aid Kit - General Purpose SRU-31/P (modified)	0.45 kg
Water Storage Bag - Sealable 5 qt. capacity	0.23 kg
Water Purification Tablets	0.03 kg
Hood & Mittens	0.28 kg
Extra Batteries - (1) for each device	0.12 kg
Magnesium Fire Starter	0.08 kg
Sun Block - Odorless	0.06 kg
Raft Repair Kit	0.20 kg
Compass - Improved SILVA 27 Type	0.08 kg
Sunglasses	0.11 kg
Head Netting	0.03 kg
Saw - Wire Type	0.06 kg
Bailing Sponge	0.05 kg
Cord - (50 ft) 500 lb Test	0.10 kg
Tropical Ration Bar (2)	0.15 kg

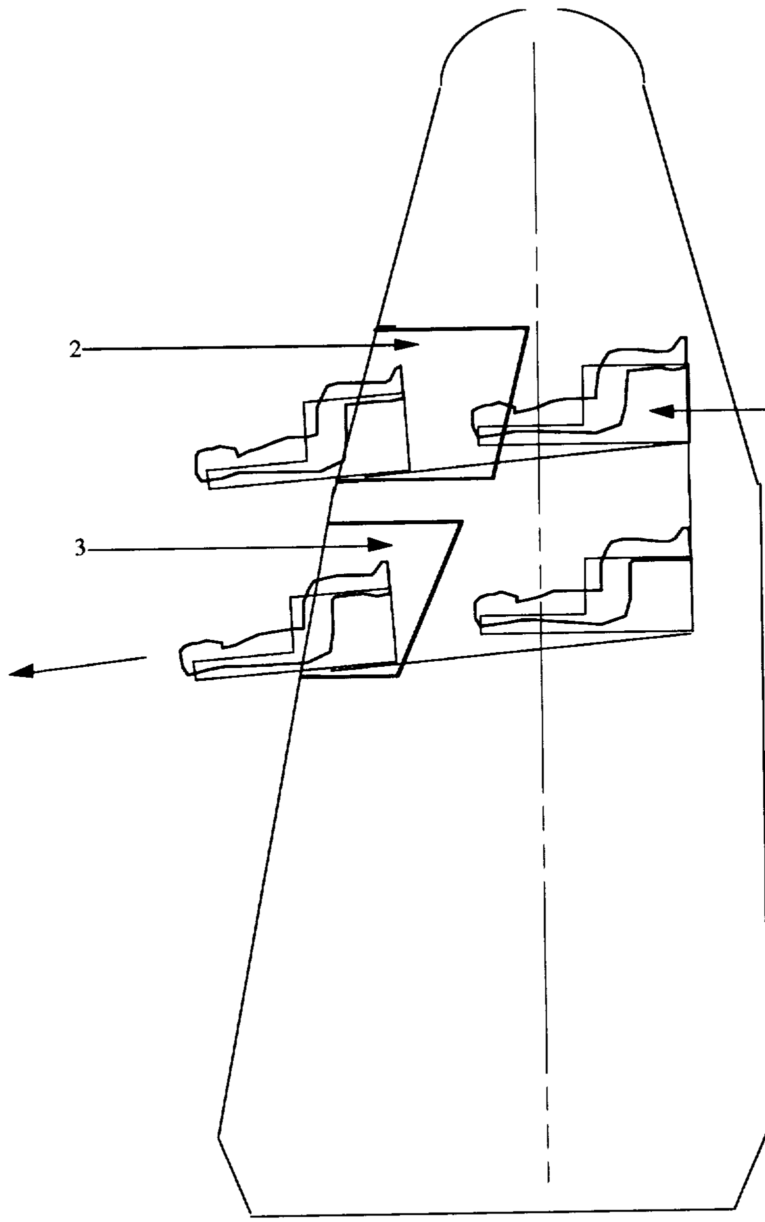
	3.95 kg

6.2.5.2 Escape Hatch Design

To incorporate an ejection seat abort system into the CM design, escape hatches using explosive bolts to separate themselves from the module are needed. The CM also needs a egress hatch for the crew to enter the vehicle as well as exit onto the lunar surface. The design of these escape hatches and the egress can be combined into one system for the CM design. It can also be considered as two separate systems.

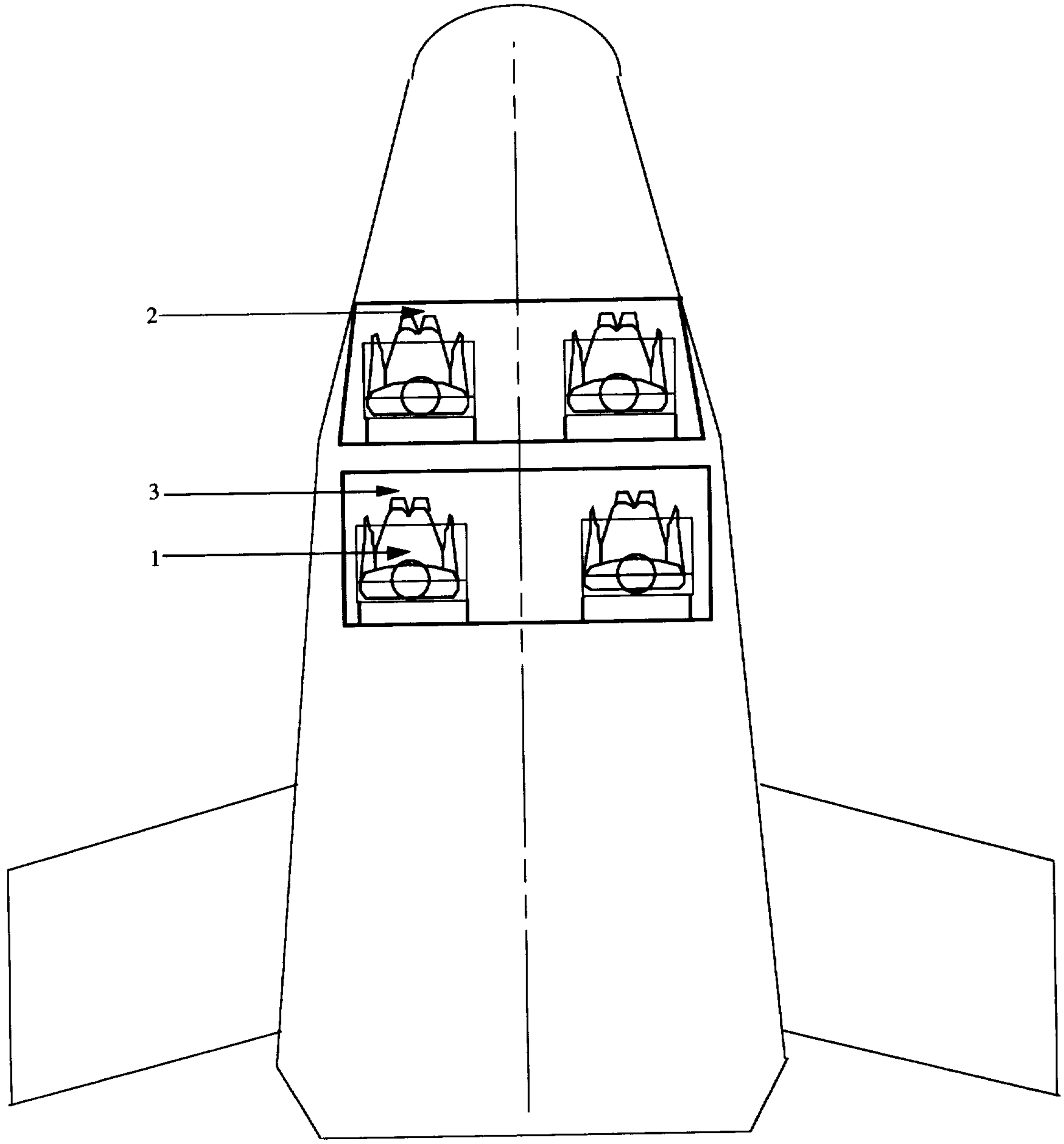
The two design options that were investigated for a combined egress/escape system are 1) a single hatch design and 2) a two hatch design. In the single hatch design, the one hatch serves as the egress and the escape hatch, and all four crew members eject out of the one hatch. In the two hatch design, one of the hatches serves as an escape hatch and the main egress, and two crew members eject out of each hatch. There was only one design option using a separate egress and escape system. This is the "hatch within a hatch" design. This uses a single hatch as an egress, which is contained within a large panel of the CM shell. This large panel can be blown off during an ejection seat abort, and all four crew members eject out of the one hatch.

The two hatch design was chosen to be implemented into the CM design along with the ejection seat units. It is shown in Figure 6-10 (a) through (d). The two hatch design does not endanger the structural integrity of the ship as the larger hatches of the other designs would. It also does not completely expose the habitable volume inside the CM in the event that the rear egress hatch will be left open, reducing the level of contamination in the habitable volume of the CM. The hydraulic or motorized systems to open the rear egress hatch are smaller than those needed for a larger single hatch design, and in general, the two hatch design is lighter and simpler.



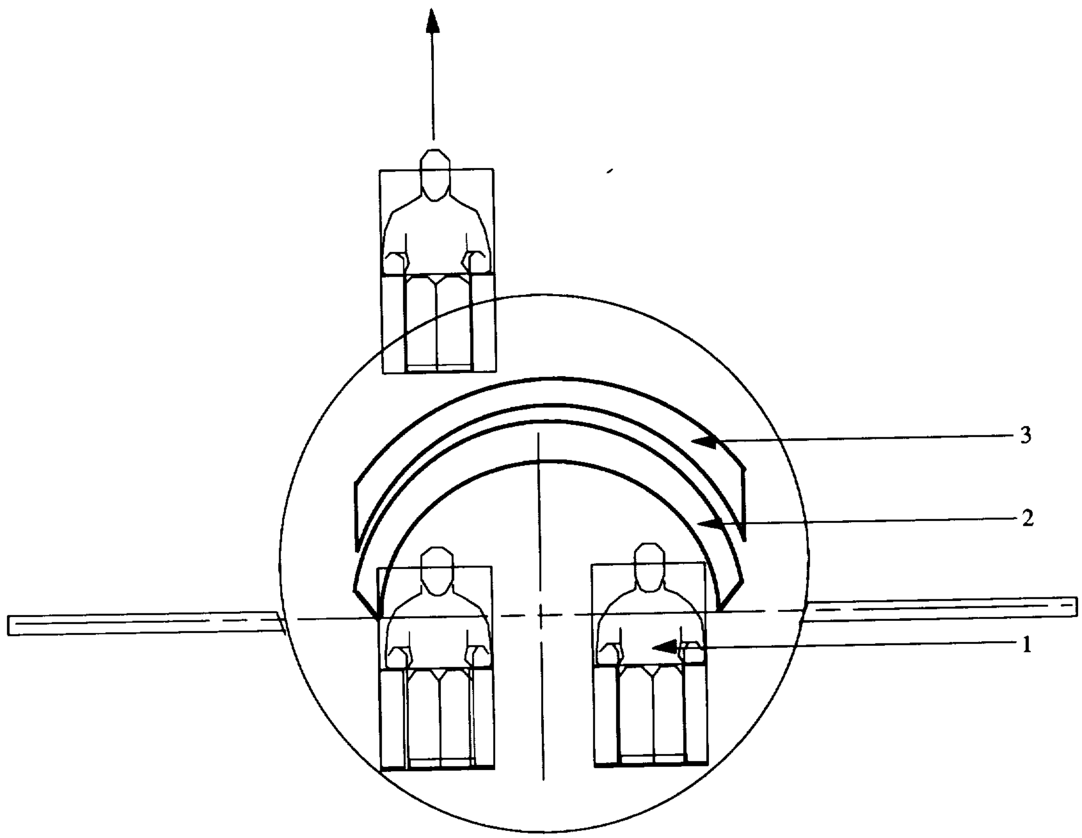
RIGHT SIDE VIEW	1 of 3
1 Ejection Seat Unit and Guide Rails 2 Front Escape Hatch 3 Rear Egess/Escape Hatch	

Figure 6-10 (a)



TOP VIEW	2 of 3
1 Ejection Seat Unit and Guide Rails	
2 Front Escape Hatch	
3 Rear Egess/Escape Hatch	

Figure 6-10 (b)



<u>FRONT VIEW</u> 3 of 3

1 Ejection Seat Unit and Guide Rails
2 Front Escape Hatch
3 Rear Egess/Escape Hatch

Figure 6-10 (c)

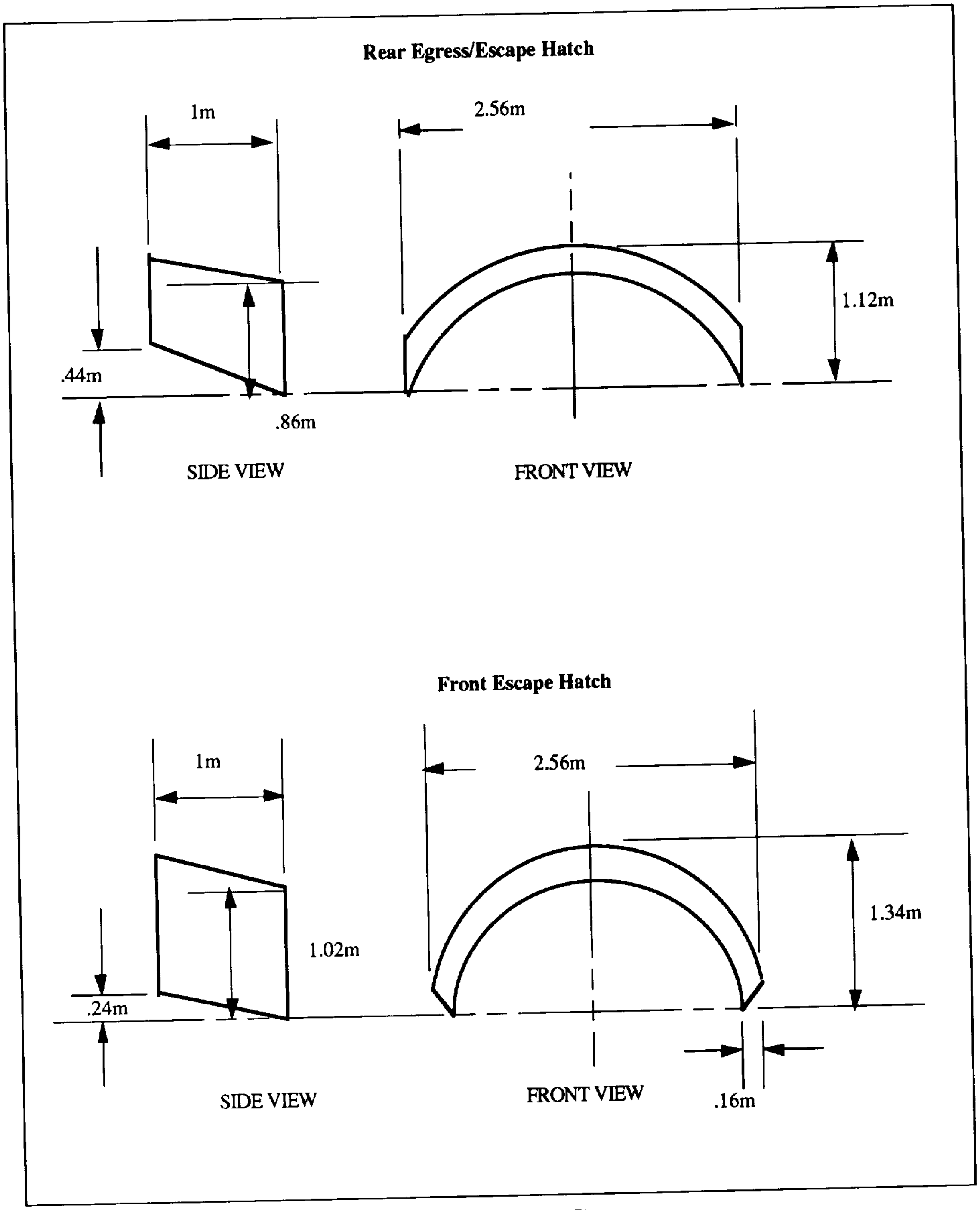


Figure 6-10(d)
Two Hatch Design

The egress assistance system for the CM uses an automated pulley to raise and lift the crew to and from the lunar surface. This system provides the ability to lift an injured crew member directly into the CM where a ladder assisted system could not. The ERM is 9.97 m high with a diameter of 6 m. The base of the egress hatch is 3 m above that height. Since there exists the risk of falling and experiencing life threatening casualties, using an automated pulley is the safest method of assisting the crew up and down the 12.97 m distance. A fold-up ladder kept in the habitat could serve as an alternate method of returning to the CM interior. The egress assistance system using the automated pulley is specified in Section 6.2.6.

Figure 6-10 shows the the CM two hatch design. The front hatch over the ejection seats of the mission commander and co-pilot serves only as an escape hatch. It releases from the CM using explosive bolts. The rear hatch over the other two crew member's seats serves as an escape hatch and an egress hatch to the surface. The automated pulley attachable to the rear hatch is stored in an accessible crew support systems storage bay. The hatches provide a 1m wide clearance for each of the crew member ejection seat units.

The design does not require a large increase in the structural mass of the CM. The distance between the two panels is 0.25m. In this space there is a main ring support which helps the overall structural integrity of the design, and enables the CM to survive the launch loads without failing. The STS Orbiter uses a similar system for the mission commander and co-pilot seats. The rear hatch also provides for ample space to egress from the CM both on the Moon and the Earth.

6.2.6 Egress System Design

6.2.6.1 Crewmember Lunar Surface Egress

6.2.6.1.1 Lunar Surface Egress System Description

The Lunar Surface Egress System (LSES) consists of a harness and winch system with a 1.5 meter long deployable boom integral to the crew hatch. The system will be driven by two independently powered electric motors each capable of lowering or lifting 250 kg to or from the surface. The winches will be controllable remotely through the Crew Module's data and control bus, or directly with a deployable control box (DCB) connected via electronic cable to the winch switches. Two Kevlar cables will be used to lower the astronauts. See Figure 6-11 for a depiction of the LSES.

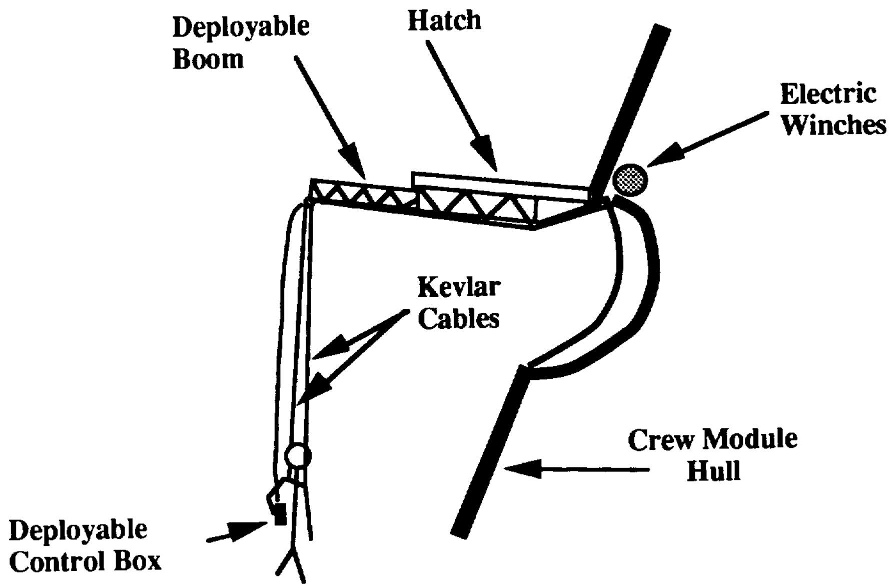


Figure 6-11
Lunar Surface Egress System

6.2.6.1.2 Exit From Capsule

When exiting the capsule on the lunar surface it will be necessary to evacuate the cabin atmosphere. This will be done by opening a pressure release valve located on the crew hatch. This procedure requires that the astronauts don the full IVA suit, including the helmet, environmental control unit, and overgarment. All exiting astronauts must also put on their Lunar Egress Harnesses (LEHs).

Once the cabin has been evacuated, the hatch will open upwards and latch into place. The crewmember who is currently exiting the CM will attach both cables and the DCB to his or her LEH. Once this is done and checked, the crewmember will sit on the edge of the hatch and activate the deployable LSES boom. As the boom deploys the astronaut will be gently suspended out from the CM. When the boom has fully deployed and suspended the crewmember beyond the ERM, the crewmember will activate the winches via the DCB and begin the descent. Upon reaching the lunar surface, the crewmember will shut off the winches, detach the DCB from the LEH, detach the LEH from the cables, and signal "all clear". The next crewmember to exit will retract the cables and the deployable boom and repeat the procedure.

6.2.6.1.3 System Disposition During Stay

When all landing operations have been completed, the LSES will be signaled via the CM control and data bus to retract the cables, DCB, and boom, and seal the hatch.

6.2.6.1.4 Return to Capsule

When it is necessary for the crewmembers to return to the CM, the LSES will be signalled via the CM control and data bus to open the hatch, extend the boom, and lower the cables and DCB.

The first crewmember to return to the CM will attach both cables to his or her LEH, check the attachments, and attach the DCB to the LEH. The crewmember will then activate the winches and begin the ascent. Once the hatch is reached, the crewmember will activate the deployable boom and slowly approach the edge of the hatch. When the boom has fully retracted, the crewmember will safely enter the CM and detach the cables and DCB from the LEH. This crewmember will then deploy the boom and lower the cables and DCB to the next crewmember.

Once all crewmembers have returned to the capsule, the boom will be stowed, the hatch sealed, and the cabin repressurized. The crewmembers will then be able to remove their helmets and breathe normally.

6.2.6.1.5 Redundancies and Safety Features

The LSES is a very important system. As such, all components except the deployable boom structure will be redundant or have replacements available. The winch motors will each be capable of lifting 250 kg, and each kevlar cable will be tested to at least 1000 kg. Both motors and cables will be attached to the same spindle, and will thus be operable in any combination. The motors will each be powered and controlled separately and with triple redundancy. An emergency braking system will also be included.

6.2.6.2 ERM Payload Lunar Surface Delivery

6.2.6.2.1 Lunar Surface Delivery System Description

The Lunar Surface Delivery System (LSDS) consists of an autonomous multiple cable and winch system with a one meter long deployable boom integral to the payload hatch. The system will be actuated by two small independently powered electric motors. The motors will be controllable remotely through the Crew Module's data and control bus. Kevlar cables will be used to lower the payload units. See Figure 6-12 for a depiction of the LSDS.

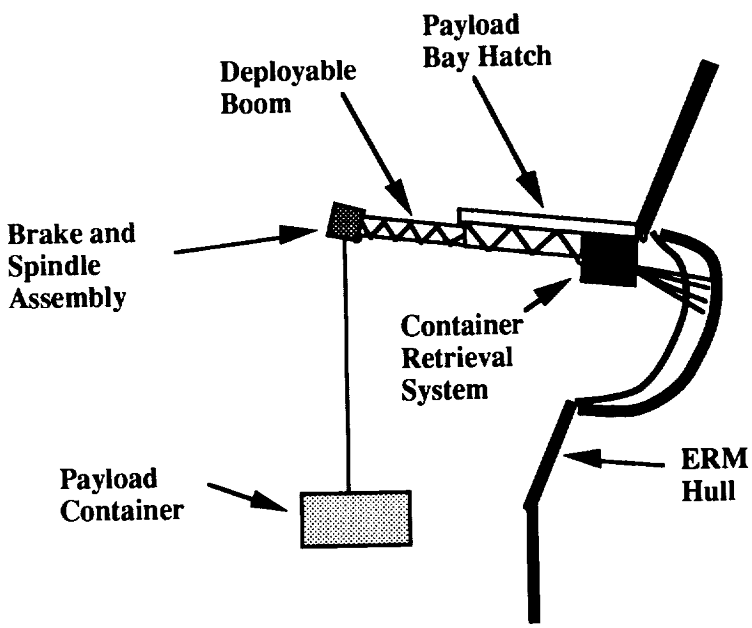


Figure 6-12
Lunar Surface Delivery System

6.2.6.2.2 Payload Stowing System Description

A Payload Stowing System (PSS) will also be used to help in delivery of the payload. The PSS consists of ten payload containers, each weighing no more than 250 kg, attached to guiderails which are integral to the ERM payload bay floor. The rails and payload containers will allow for easy removal of the payload from the ERM payload bay. Each payload container will be no more than 0.5 meters high and 1 meter in width and depth. See Figure 6-13 for a depiction of the PSS.

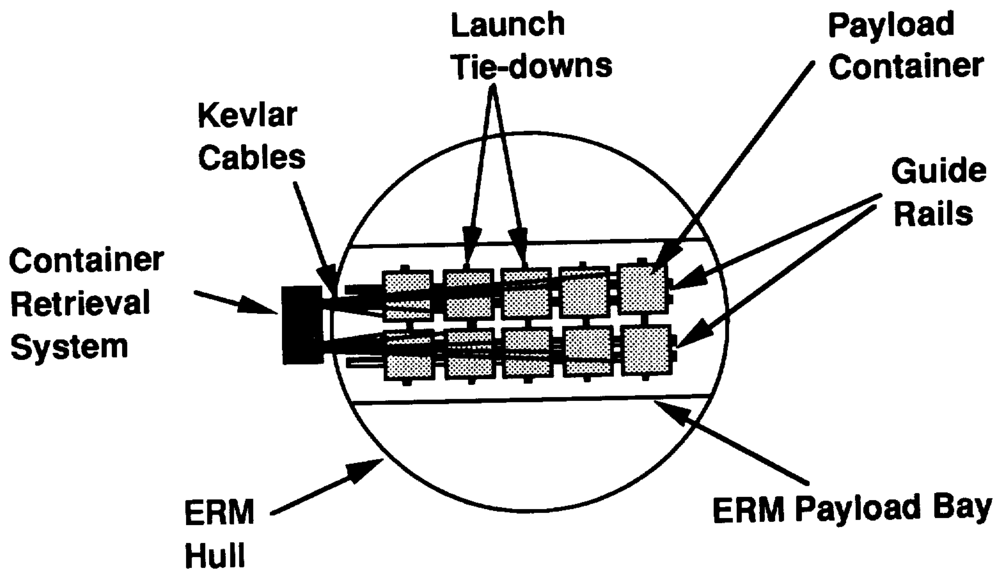


Figure 6-13
Payload Stowing System

6.2.6.2.3 Procedure for Delivery

When the LSDS is first activated by the crew, the payload hatch will open and the launch tie-downs on the payload containers will release. A Container Retrieval System (CRS) will allow the electric motors to pull the first payload container into position at the edge of the hatch by winding an attached cable around a cable spindle. The spindle will be moved into place on the LSDS boom, and a mechanical brake will attach to the spindle. The LSDS boom will then deploy out from the ERM, suspending the payload container over the lunar surface. A control system will slowly release pressure on the brake until the payload container begins to descend to the lunar surface. The control system will maintain a slow rate of descent until the container reaches the surface.

Once the container reaches the surface, the LSDS will await a signal from the crew. Once the crew signals "all clear" the LSDS will release the spindle and cable from the boom, allowing it to fall to the surface. The LSDS boom will then retract, and the CRS will pull the next container into position. The LSDS will await a "start" signal from the crew before beginning the process again. This process will continue until all payload has been delivered to the lunar surface.

6.2.6.3 Crewmember Terrestrial Egress Procedure

Upon landing on the Earth's surface the capsule will remain sealed for approximately 20 minutes to allow for cooling and give time for outgassed gases to dissipate. A surface vehicle will then roll into position, and the hatch will open. A boom from the surface vehicle equipped with a personnel holder will be lowered into the hatch to remove the crew.

In the event of a fire or other accident on the ground, the ejection seats will be utilized.

6.2.7 Reentry Procedure

6.2.7.1 Detachment from ERM

See Volume 1, Sections 5.1.3.4 and 5.2.2.2 and Volume 3, Section 6.2.2.1 for information regarding the interfaces of the CM and ERM and how detachment occurs. See also Volume 1, Section 5.3.7.2 for the CM mission profile upon reentry of the Earth's atmosphere.

6.2.7.2 RCS Stage

The Reaction Control System on the CM will maintain its stability once it has detached from the ERM and before the stabilizing fins are deployed. See Section 6.2.1 for details about the location, masses, and thrusts of the RCS on the CM.

6.2.7.3 Stabilizing Fin Stage

A description of the stability, L/D, and wings of the Crew Module can be found in Volume II, section 2.2.4.

6.2.7.4 Drogue Decelerator

The primary function of the drogue parachute is to decelerate the craft to a dynamic pressure of 2390 N/m^2 at an altitude of 3000 m. At this location the dynamic pressure is low enough to deploy the primary recovery system, the parafoil.

6.2.7.4.1 Requirements for the Drogue Decelerator

The following requirements have been established for the drogue parachute:

1. Decelerate the reentry vehicle to a dynamic pressure of 2390 pascals at 3000 m (a velocity of 72 m/s).
2. Have G-loads under 3.0g.

6.2.7.4.2 Drogue Decelerator Options

The decelerator drogues studied were: conical ribbon parachute, hemispherical ribbon parachute, and ballute. Table 6-4 gives the basic characteristics for each parachute: the mach number range and the coefficient of drag and opening force [Minnesota 91].

Table 6-4: Drogue Characteristics

TYPE	Mach Range	Cd	Cx
Conical Ribbon	0.1 - 2.0	0.50 - 0.55	1.05 - 1.30
Hemispherical Ribbon	1.0 - 3.0	0.30 - 0.46	1.00 - 1.30
Ballute	0.8 - 4.0	0.51 - 1.20	1.20

The study of these decelerators resulted in the choice of the conical ribbon design based upon its high coefficient of drag ~ .5 and its effectiveness at the lower mach number range of parafoil deployment (M = .22). The ballute parachute offers higher drag but is not effective at low velocities.

6.2.7.4.3 Drogue Design

Using the design method outlined in the "Recovery Systems Design Guide" the following bi-conic ribbon parachute was designed. Table 6-5 gives the pertinent parameters.

Table 6-5: Bi-conic Ribbon Parachute Design

Design Parameter	Value
Diameter, Dc	4.48 m
Vent Diameter, Dv	0.30 m
Line length, le	6.33 m
Height, hp	3.5 m
Number of Gores	24

Component	Material	Weight
Horizontal Ribbons	MIL-T-5608 E III	20.0 kg
Suspension Lines	MIL-T-5608 E VI	5.6 kg
Total		25.6 kg

The decelerator is reefed prior to the full deployment. The reefed area has diameter of 1.5 m with a total area of 6.9 m^2. The reefed drogue is deployed at 27,000 m at a velocity over 340 m/s. The reefed stage lasts for 55 seconds. The stage is disreefed once the expected loading of the full drogue is less than 1g. The drogue has a total area of 63.37 m^2 once fully deployed. See Figure 6-14.

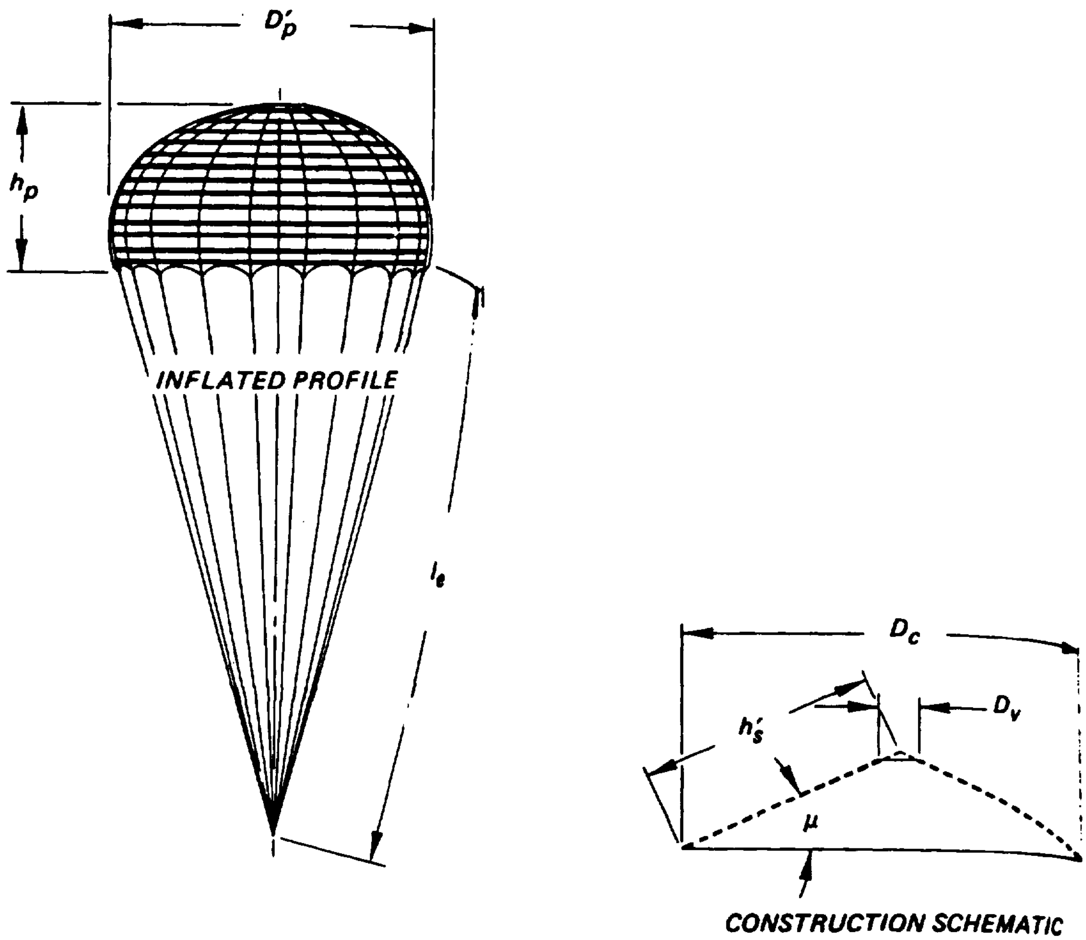


Figure 6-14
Construction of Drogue Parachute
[Ewing,Bixby,Knacke, December 1978]

6.2.7.4.4 Drogue Performance

The performance of the drogue parachute is shown in Figures 6-15 and 6-16.
The reefing behavior can be observed by the initial deceleration of the craft as shown in Figure 6-15. Once disreefed, the craft is decelerated to 72 m/s at 3000 m. The entire process takes 550 seconds as displayed in Figure 6-16.

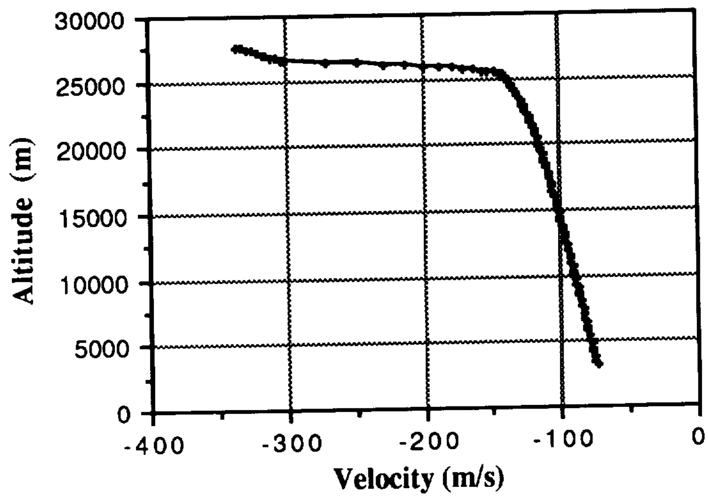


Figure 6-15
Altitude vs Velocity for the Drogue Decelerator

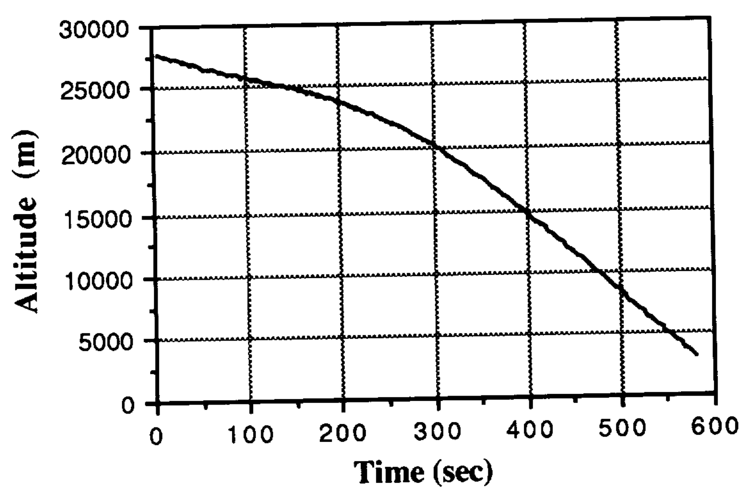


Figure 6-16
Altitude vs Time

6.2.7.5 Parafoil Stage

6.2.7.5.1 Overall Shape and Dimensions

The Ram-Air Parafoil (RAP) is a hybrid of a parachute and an airfoil and will be used in the last stage of reentry to Earth - the final 3000 m until touchdown. The parafoil is stored and deployed exactly like a conventional parachute and when fully deployed looks like a low aspect ratio wing. It is made entirely of fabric, containing no rigid, plastic members. A frontal and profile view of a general parafoil are shown in Figure 6-17.

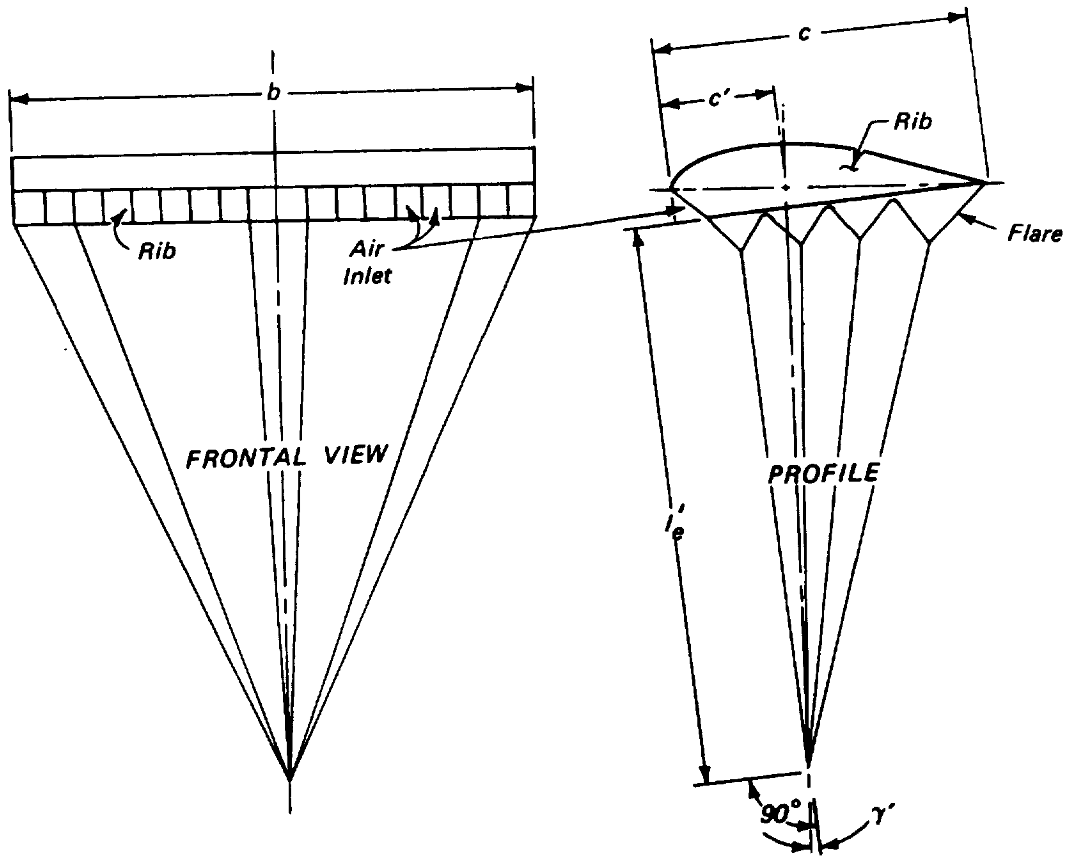


Figure 6-17
General Parafoil Design

As shown in Figure 6-17 Frontal View, fabric ribs divide the parafoil into many box-type airfoil-shaped compartments, called cells. These exist spanwise across the parafoil and transmit air pressure between them by numerous openings in the ribs. To allow the ram air pressure to expand and maintain the shape of the parafoil, the leading edge is open along the entire span as seen in Figure 6-17 Profile. Pressure equilibrium is maintained by the parafoil cells at all times, which creates a very stable and reliable system. Although parafoils do not utilize all the material as lifting surfaces, this negative aspect is overridden by the excellent aerodynamic shape.

The basic RAP configuration used for Project Columbiad is displayed in Figure 6-18. It has a total wing planform area of 568.5 m^2 when fully deployed and is disreefed in three stages. In other words, it will expand up and outwards at three different points before the final area is achieved. With a chord of 16.86m and a final span of 33.72m, its final aspect ratio will be 2.

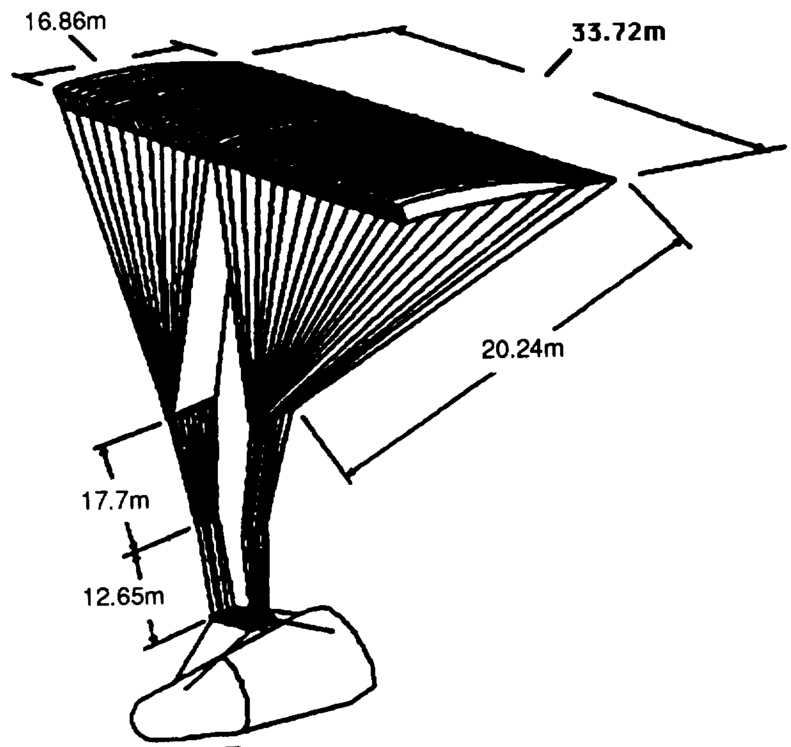


Figure 6-18
Parafoil Configuration

For a more detailed breakdown of the three disreefing stages, their dimensions and characteristics, see Table 6-7 in Section 6.2.7.5.3.

6.2.7.5.2 Airfoil Shape/Design

The airfoil chosen for our RAP was a modified NACA 2210 airfoil. It is an extremely efficient airfoil which was chosen for its outstanding aerodynamic characteristics. For more information about the NACA 2210 see Volume 2, Section 2.2.4.1.3. The modified NACA 2210 with actual dimensions is shown in Figure 6-19.

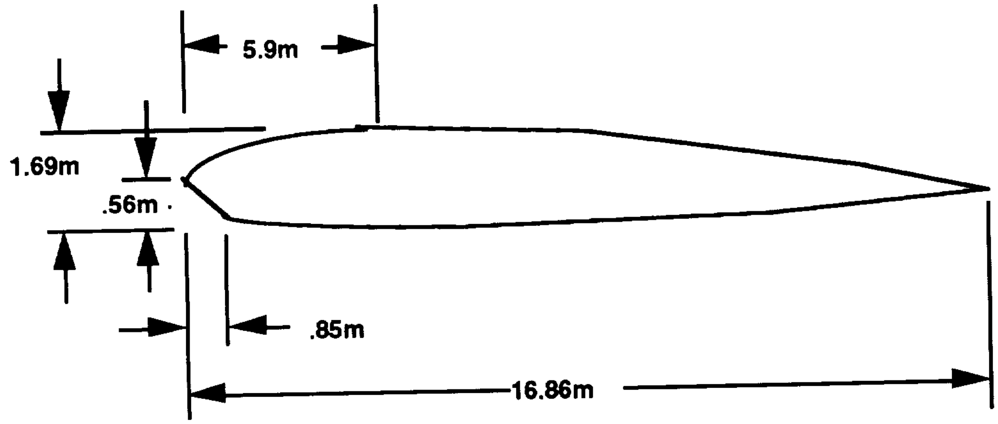


Figure 6-19
Modified NACA 2210 Airfoil

The leading edge was opened to allow for the ram air pressurization which is essential for the parafoil to inflate and maintain inflation. The opening of .56m in height and .85m in length along the span was based on the ratios of existing parafoils. The loss of aerodynamic efficiency was accounted for in the aerodynamic characteristics of the NACA 2210, although the accuracy of these numbers is an area which needs more investigation if the design is to be implemented. Table 6-6 below shows the modified aerodynamics coefficients.

Table 6-6: Modified NACA 2210 Characteristics

alpha = 0°
C_L = .210
C_M = -0.035
C_D = 0.00611
L/D = 34.46

6.2.7.5.3 Deployment/Reefing

Initial deployment of the parafoil requires a small, 1 m pilot chute which is shot out once the storage container is opened. Since the parafoil is stored in a .57m^2 volume in the front of the CM, one of the main cables of the parafoil must be connected to the rear of the CM. This is required to dissipate the large loads which the structure will experience once the first stage of the parafoil inflates. In addition, the connection to the rear of the CM is needed to prevent the CM from overrotating from it's nose-down drogue parachute configuration to the horizontal parafoil configuration. See Figure 6-20 below which shows the parafoil release procedure.

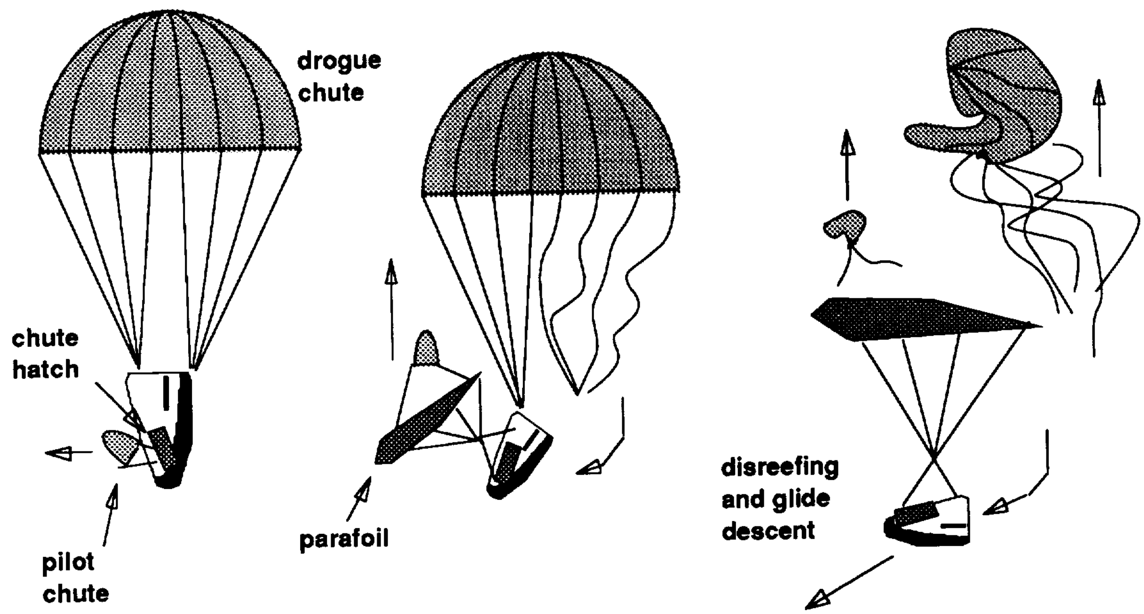


Figure 6-20
Initial Parafoil Release Procedure

After the drogue chute has decelerated the CM to 73 m/s at a height of 3048 m, a small hatch in the front of the CM is shot off, followed by a pilot chute which pulls out the parafoil. Since the CM is nose-down at this point, the two main cables of the parafoil comes into the picture. One of the main cables has been led outside the CM to a ring near its rear and then back into the parafoil storage container for the duration of the mission. When the parafoil is pulled out, the cable becomes taught and allows the force to be divided among the back and front of the CM. See Figure 6-21 below for a visualization of this loop-like layout.

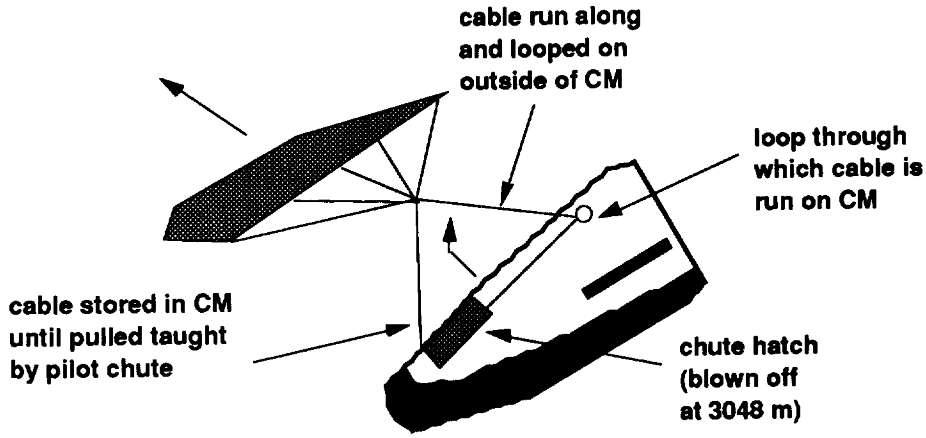


Figure 6-21
Parafoil Cable Deployment Detail

The parafoil becomes centered above the middle of the CM because the first main cable, which lined the outside of the CM, and the second main cable stored in the CM are of equal length. As the first stage of the parafoil becomes inflated, the two cables become taught and prevent "parafoil pitchover" due to the airfoil lift and do not allow the momentum of the CM rear to cause overrotation. During the rotation, the drogue chute and pilot chute are released by pyrotechnic cutters and the CM attains the necessary angle for the parafoil stage.

Since it would be quite difficult to deploy a 568.5m^2 parachute in one stage without creating enormous loads and g-forces, we must inflate the parafoil in stages. Using a parafoil technique known as mid-span reefing, the outside cells are inflated first, followed by the middle cells as seen in Figure 6-22 below.

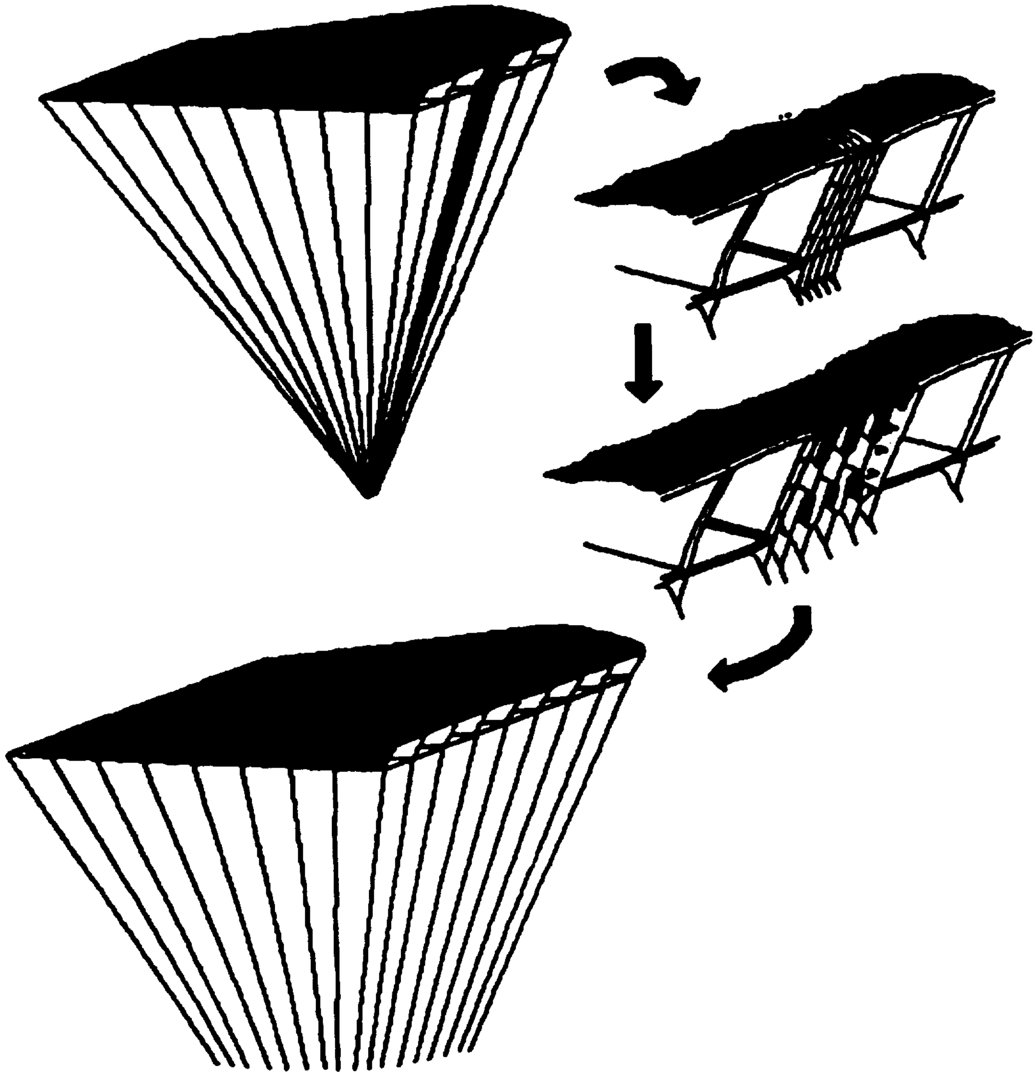


Figure 6-22
Mid-Span Reefing Technique

In the deployment of the first stage, the outer cells are able to inflate due to the ram air pressure while the second and third stage cells are folded and stored in the middle of the first stage inflated cells. They are laced into place to the adjacent inflated cells and once a pyrotechnic cutter severs the locking ties, the stowed cells are released and are able to inflate. Thus, two disreef commands will occur during the deployment of the parafoil. Exact dimensions, reefing ratios, and load factors for each of the three stages are given below in Table 6-7.

Table 6-7: Deployment/Reefing Data

	1st STAGE	2nd STAGE	3rd STAGE	FULL OPEN
Delta Area (m^2)	142.2	142.1	284.2	-
Total Area (m^2)	142.2	284.3	568.5	568.5
Delta Span (m)	8.43	11.8	13.49	-
Total Span (m)	8.43	20.23	33.72	33.72
Chord (m)	16.86	16.86	16.86	16.86
Aspect Ratio	.5	1.2	2.0	2.0
Delta Cells	6	8	10	-
Total Cells	6	14	24	24
Reefing Ratio	.25	.25	.5	1.0
Load Factors (N)	93741.7	42332.6	37502.9	-
Delta Gore Length(m)	12.65	17.7	20.24	-
Total Gore Length(m)	12.65	30.35	50.58	50.58

As seen from above, the chord remains constant because both inflated and stowed cells are the same length. But the aspect ratio changes from .5 to 1.2 to 2 for the fully inflated parafoil because the span is increasing with each stage. The total number of cells inflated is 24, another characteristic which was based on ratios from existing parafoil designs. It is also worthwhile to note that the initial stage takes the greatest load, nearly 10,000 N, while the second and third stage experience loads near 4,000 N. Thus, the gores (tethers connecting the parafoil to the CM) must be able to withstand a maximum of more than 10,000 N at any single time and a constant force of more than 4,000 N for the duration of the parafoil use.

Although not shown on any tables, there are a few other important characteristics of the parafoil which need to be mentioned. Firstly, the parafoil is designed to have 48 gores, a number which will allow the huge deployment loads to be absorbed by the CM. Secondly, the parafoil must be stored in the CM for the duration of the flight, so the volume and mass are minimized as much as possible. A total parafoil weight (including gores) of 45.84 kg will be compacted into a .57m^2 compartment in the front end of the CM.

6.2.7.5.4 Trajectory

The initial deployment of the parafoil's first stage will occur at an altitude of 3048 m where the vertical velocity of the CM will be 72.85 m/s. Within the next 44.5s, the parafoil's second and third stages will be inflated to achieve full wing planform area. Thus, at an altitude of 1829 m, the parafoil will be at full capacity, with an lift-to-drag ratio of 13.57. A complete list of trajectory data is shown in Table 6-8.

Table 6-8: Trajectory Data

EVENT	ELAPSED TIME (s)	ALTITUDE (m)	VERT.VEL. (m/s)	HORIZ. VEL. (m/s)
Deploy Parafoil	0.0	3048	72.86	0.0
Disreef 2nd Stage	5.6	2743	36.58	107.82
Disreef 3rd Stage	26.6	2134	21.34	72.39
Flap Release	44.5	1829	12.80	43.43
Full Glide	59.9	1676	7.04	65.15
Flare Maneuver	419.6	31	2.11	39.09
Touchdown	436.5	0	1.50	20.24

The maneuvers following the disreefing to full area are based on the University of Minnesota report, under the assumption that all parafoils follow essentially the same manuever procedure when landing on the Earth. Upon touchdown, the parafoil is detached with pyrotechnic cutters so the landing gear can use its brakes to safely bring the CM to a stop.

6.2.7.5.5 Guidance and Control

The parafoil control system is based mostly on the deflection of the trailing edge. Support cables led along the trailing edge gores can be reeled in by use of electric motors, causing a flap deflection of up to 20°. This deflection is incurred upon the 4.21m (one quarter of the chord) at the rear of the parafoil and is displayed in Figure 6-23.

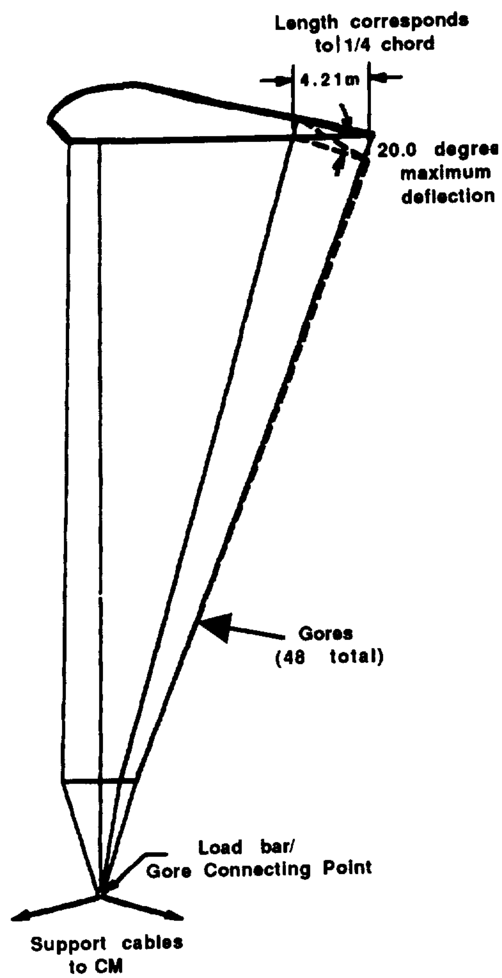


Figure 6-23
Parafoil Control System

Much like the ailerons of a wing, the trailing edge is divided into two parts and the deflection can be performed on either or both sides of the trailing edge. This allows turn control and easy stabilization for the parafoil. Of course, since the trailing edge can be pulled down a maximum of 20°, we must have a short layer of fabric which is folded up along the quarter chord point and can be extended when the flap is pulled down. This can be seen in Figure 6-24.

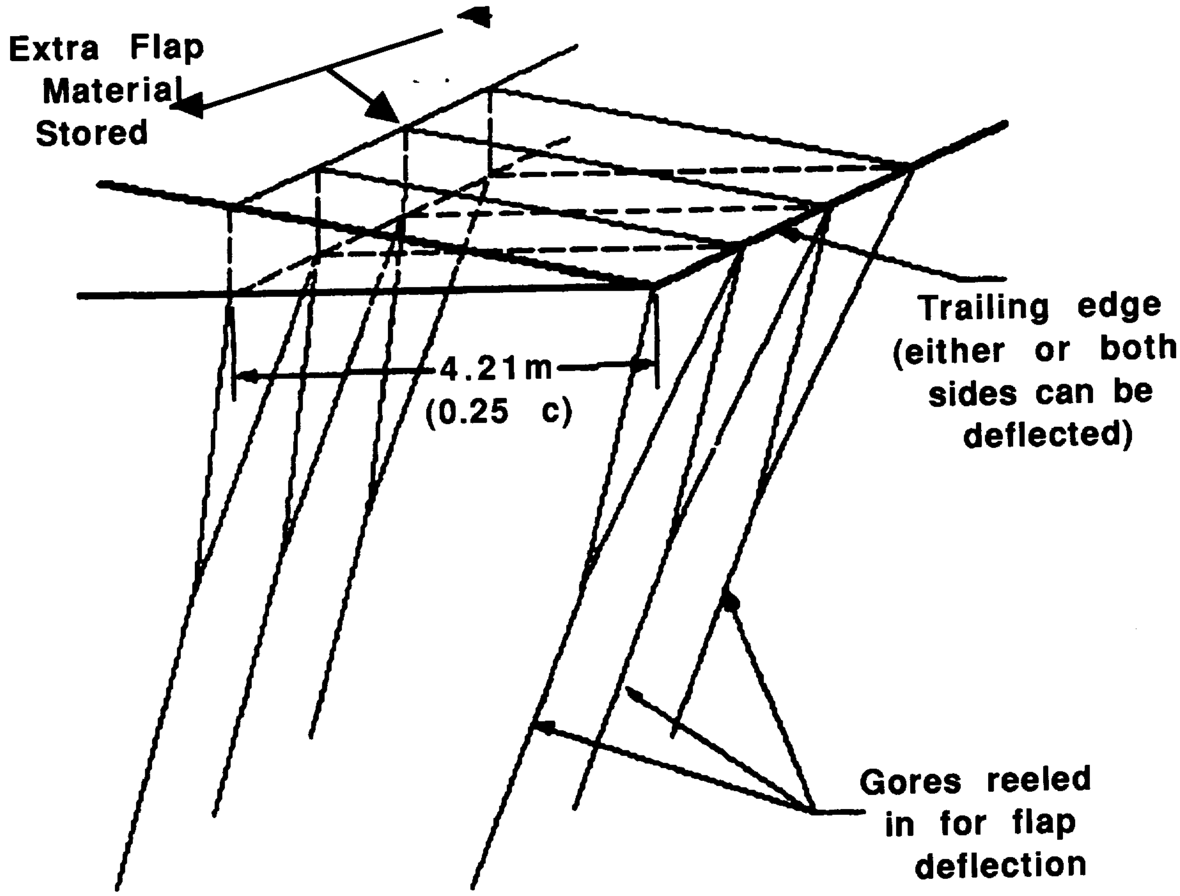


Figure 6-24
Trailing Edge Deflection Detail

Both the flap release and flare manuevers specified in the trajectory procedure of Section 6.2.7.5.4 utilize the trailing edge deflection capability. The flaps are at 20° deflection during the disreef staging because "parafoil pitchover" can be caused by a sudden burst of lift and this configuration will reduce the seriousness of this problem. The lift produced by the flap deflection will ease the rapid increase of lift as the cells begin to fill with air and the parafoil takes shape. As more of the parafoil is inflated, the lift increases and the flap deflection is decreased to 10° to lessen the lift it is producing. Shortly after the third stage is completed, the flap deflection should be gradually decreased to 0° in order to decrease the drag caused by the flaps. This is denoted by "Flap Release" on Table 6-y+2 in Section 6.2.7.5.4. Once the flaps are at 0° deflection, the stage is denoted by "Full Glide."

The flaps are once again used during the flare manuever shortly before touchdown. The flare will reduce the vertical velocity to 1.5 m/s, a reasonable landing speed for the CM's landing gear system and the horizontal velocity of 20.24 m/s. Even more importantly, the flare manuever will orient the CM into an acceptable position for landing. The flare is achieved by deflecting the flaps while using a pyrotechnic cutter to release a lazyleg cable. This lazyleg causes the cable length to the gore center to be increased, which takes the potential energy of the CM tail to rotate and push the nose upward. Below in Figure 6-25, the flare maneuver procedure is pictured.

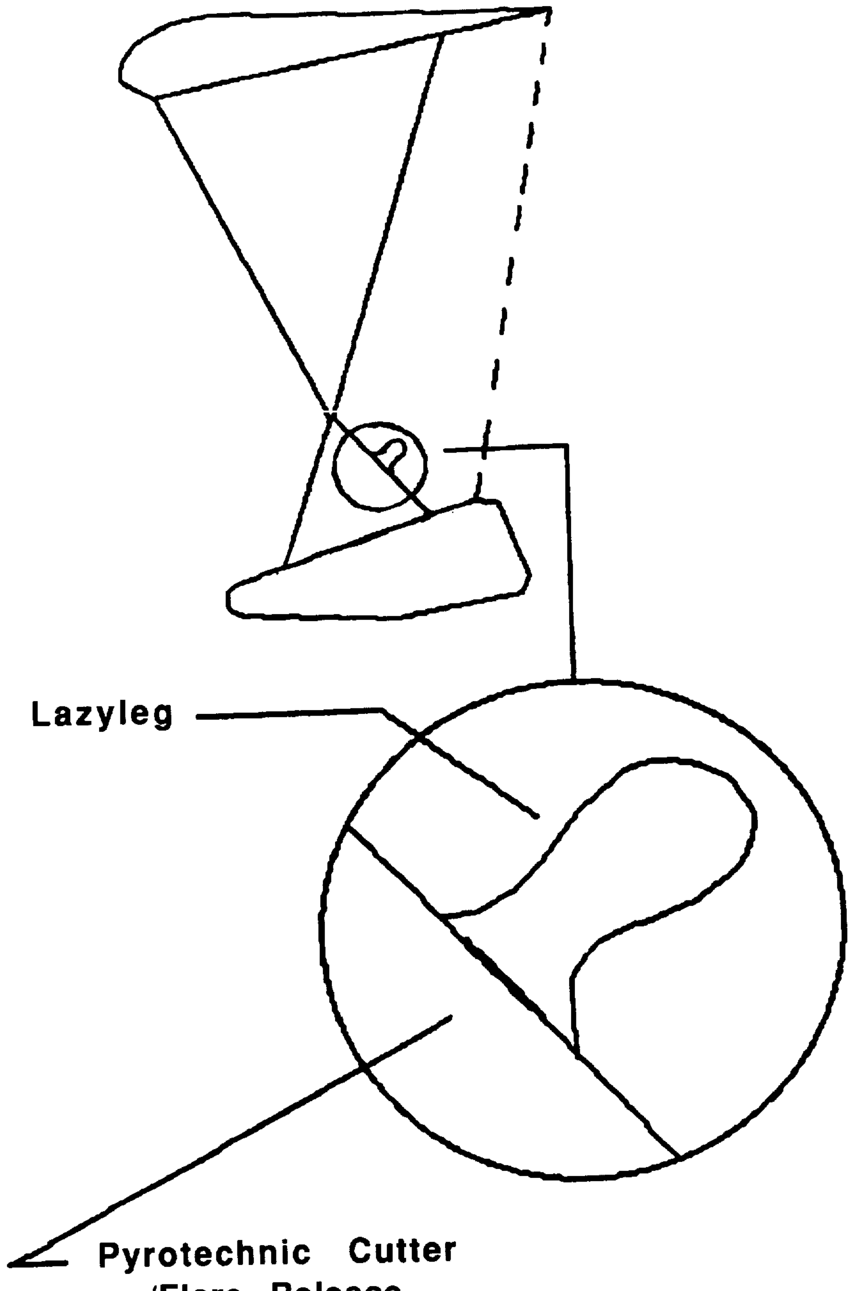


Figure 6-25
Flare Maneuver Detail

The amount of flap deflection depends upon the effect of the lazyleg, an issue which needs more investigation if this is to be used. Based on the University of Minnesota report, this flare technique is a representation of an existing parafoil procedure used immediately before touchdown. The final landing configuration is shown below in Figure 6-26.

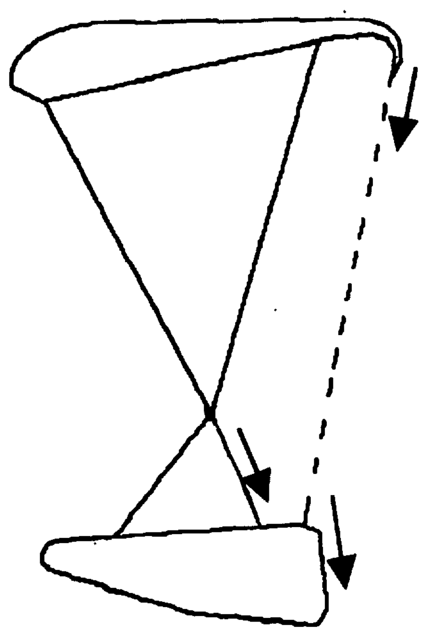


Figure 6-26
Touchdown Configuration

6.2.8 Instrument and Control Panel

The Instrument and Control Panel (ICP) design is intended to provide timely and quickly understood information to allow for easy control of the spacecraft, efficient troubleshooting, and psychological relief for the astronauts. This is accomplished through display of real-time video images of the exterior of the spacecraft, quickly interpreted data on the status of the spacecraft systems, and efficient interfaces with the control and data bus of the spacecraft.

The ICP layout includes two identical control panels, one for each of the two on-duty crewmembers. The individual ICPs integrate two display screens: a cathode-ray tube Multi-Role Screen and a liquid crystal display Secondary Data Display. In addition there are various interfaces, including control balls and interface keys for cursor control and menu selection, a keyboard for data entry, and a camera control unit (CCU) for controlling the exterior cameras. See Figure 6-27 for a diagram of the ICP layout.

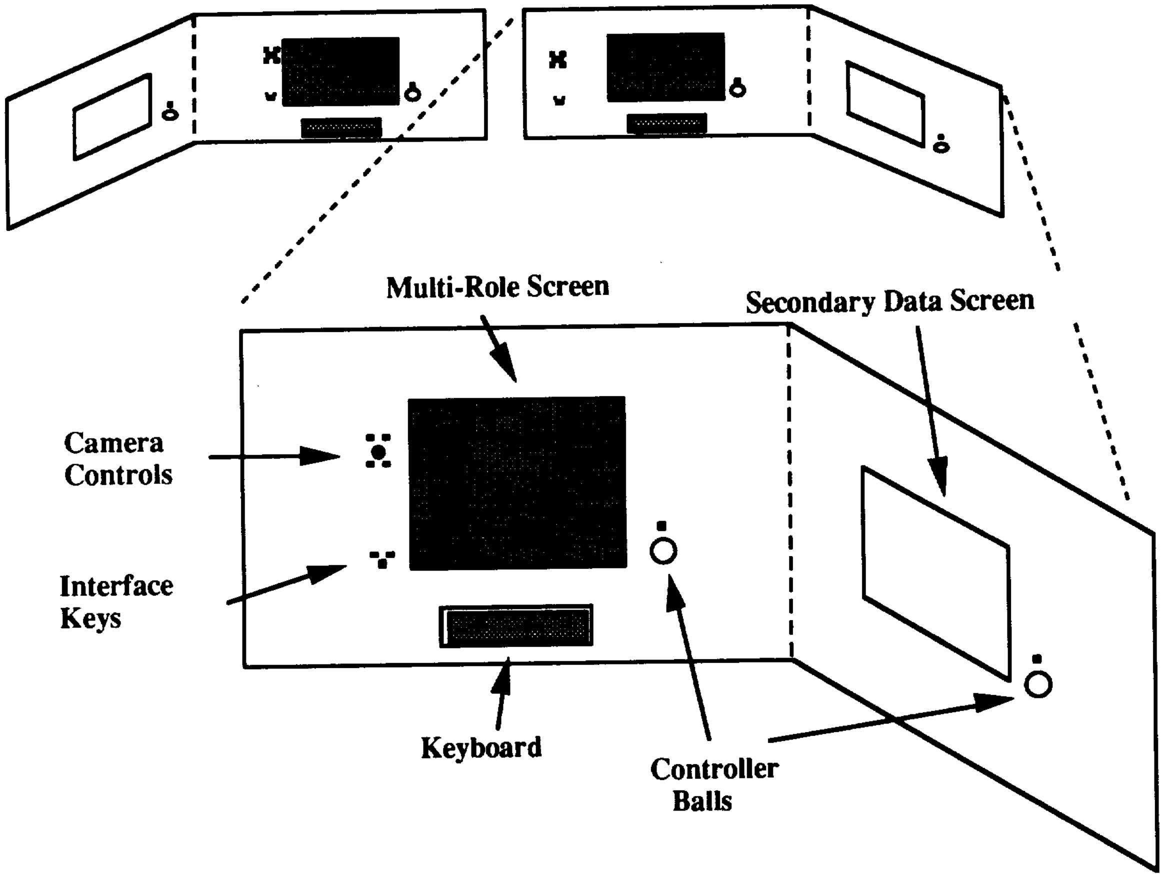


Figure 6-27
Instrument and Control Panel Layout

6.2.8.1 Display System

6.2.8.1.1 Multi-Role Screen

The Multi-Role Screen (MRS) will serve as an integrated video and data display during most modes of operation. It will be capable of clearly displaying high-definition color video, and will serve as the main control panel for the commander. The video signal will be supplied by the activated external camera. Status data will be overlaid across the top and bottom of the screen.

6.2.8.1.2 Secondary Data Screen

The Secondary Data Screen (SDS) will display more detailed textual and graphical status information, and will be the main display used for troubleshooting and during command, control, and communications interfacing. It will be able to display 256 colors with high resolution graphics for ease of data interpretation.

6.2.8.1.3 Normal Mode Data Layout

During normal mode operations the MRS configuration will consist of a video display of the activated camera with a brief summary of spacecraft status overlaid around the outside of the screen. The information across the top of the screen will be relatively rudimentary, consisting of a color-coded graphical representation of the relative status of each system (critical failure, poor, stable, and no failures corresponding to red, orange, yellow, and green). This is to give a quick, at-a-glance synopsis of all systems in order to allow the commander to have a feel for the overall status of the spacecraft and respond to problems quickly.

The bottom part of the screen will display mission level information, such as a clock, elapsed time, mission milestones, selected camera, and reminders of crew duties. A small data and query entry window will also be provided for status checks.

The SDS will be divided in three sections. The largest window will be a data retrieval window capable of displaying specific information about failures and subsystem status. A smaller window will keep a running list of failures or problems which the computer has taken care of without crew input, and a similarly sized window will show a prioritized list of failures or possible problems that require action by the crew.

6.2.8.1.4 Maneuver Mode Data Layout

During maneuvers the MRS will still display video images from the activated camera and graphical status bars across the top of the screen. However, the bottom of the screen will be devoted to various instrument readings such as orientation, rates, altitude, velocities, and fuel consumption in order to aid the pilot in whatever maneuver is being performed.

The SDS will be broken up into four equally large sections, three of them the same as the normal mode display, and the fourth devoted to additional control information.

6.2.8.1.5 Launch Mode Data Layout

During launch the MRS will be entirely devoted to graphical and textual representations of crew, stage, and capsule status, with no video display. The SDS will have a Normal Mode Configuration.

6.2.8.1.6 Display Contingencies and Redundancies

Each side of the ICP will be separately and redundantly powered and connected to the CM data and command bus. Each side will be controlled separately by the crewmember at that station. The MRS will be capable of displaying all information displayed by the SDS, and the SDS will be capable of displaying all textual and graphical information displayed by the MRS. The SDS will not be capable of displaying video images.

6.2.8.2 Data Interfaces

For the most part, human interface with the computer will be performed through the use of controller balls. A cursor on the screens will be controlled by the controller balls, and a button above the controller balls will elicit a menu from the specific site chosen on the screen. However, a keyboard will also be provided for more complicated or detailed interfacing. Several macro interface keys will be installed around the MRS and SDS for quick access to specific information and quick action in the case of emergency.

6.2.8.3 Visual Data Collection System

The Visual Data Collection System (VDCS) will provide clear external images to the crew for landing, inspection of exterior systems, docking, and psychological relief. It will do this through the use of two pairs of high definition cameras, one pair located in the nose of the CM, and the other located in the belly of the craft. The video images will be displayed on the Multi-Role Screen of the Instrument and Control Panel.

The cameras will be controlled via a joystick and activation keys to the left of the MRS. Each camera pair will have a full field of view except where blocked by the hull of the CM. The cameras will be recessed into the spacecraft body during launch and reentry in order to protect them (and the spacecraft) from heating and undue aerodynamic stress. During all other parts of the mission (including landing), the cameras will be deployed and active. Each camera will be powered separately to decrease the chance of total system failure.

During the stay on the lunar surface, the cameras can be remotely controlled to allow mission control additional images of astronaut activities on the lunar surface.

6.2.8.4 Flight Controls

Since most control of the Crew Module will be automatic, the flight controls are relatively rudimentary, and consist of a joystick integrated into the right arm rest of each front seat, a slide control integrated into the left arm rest, and various buttons and switches located around the MRS and on the keyboard.

During landing on the lunar surface, the joystick will be used to mark a landing site on the video display of the MRS chosen by the commander. The guidance system will then take control of the descent.

During landing on the Earth's surface, the joystick will be used to fly the CM like an aircraft, and the slide control will be used as an additional control for the parafoil. The guidance and control system will deal with control during reentry and maintenance of stability during flight.

6.3 Crew Systems

The crew module must provide the basic necessities of life and other necessary equipment for the astronauts. The crew systems requirements include a 99% reliability. This reliability will be achieved by having a system with a 99% reliability or by providing three levels of redundancy in the systems [Shea, 1992]. The systems that require redundancies have an individual 95% reliability and when three systems are connected in parallel then the net reliability will be the desired 99%. Crew systems also has established a factor of safety of 1.5 for all consumables. These two aspects, reliability and safety factor, affect the crew systems' drivers. The drivers are mass, volume, and power requirements.

Crew systems includes crew provisions, environmental control for the crew module, and other equipment. The budgets for these systems are presented in Table 6-9 to provide a total. Each system is completely broken-down in this section and further budget specifics are provided.

Table 6-9 :Total Budget of Crew Systems For The Crew Module

System	Mass (kg)	Volume (m3)	Power (watts)
Crew Module Provisions	588.82	11.67	0
Crew Module Environmental Control	468.94	5.923	2309.8
Crew Module Bioinstrumentation	22.98	0.08	100
Crew Module Spacesuits	120.3	2.255	0
Other Crew Module Equipment	33.5	0.12	200
TOTALS	**1234.54**	**20.048**	**2609.8**

6.3.1 Crew Provisions

The analysis for the required crew provisions for the crew module follows the same methods as the habitat. Refer to section 7.1 of Volume II for the methods used to obtain the mass, volume, and power budgets. However, there are differences between the crew module and the habitat. The crew module supplies were based on provisions for six days with a factor of safety of 1.5. Thus, the supply of clothing, food, oxygen, nitrogen, drinking water, wash water, and toiletries are based on six days. Other things to note are the medical kit and hygiene station [Joels, 1982]. Also, the pressurized volume of the crew module is 15m^3, this is used in determining the mass of cabin oxygen and nitrogen needed (Subsection 7.2.1.2 of Volume II).

Table 6-10 provides the consumables for a four person mission for six days with a factor of safety of 1.5 built-in. This factor of safety and the two extra cabin atmosphere supplies (in case of depressurization) provide more consumables than required for a four person - six day mission if everything goes as planned. The oxygen and nitrogen provide enough for thirteen days due to the extra supplies in reserve for repressurization atmosphere. However, the drinking water only lasts nine days since it only has a factor of safety of 1.5 and no reserve supplies.

Table 6-10: Crew Module Provisions

	Mass (kg)	Volume (m3)
Crew Provisions total	**588.82**	**11.67**
Crew of four	300	10
Clothing	-	0.2
Shoes	4	-
Dress (1 week)	18.4	-
Sleepers	32	1.2
Food (dry weight)	22	0.2
Medical kit	8	0.01
Oxygen		
Daily Supply	32.75	-
Cabin Atmosphere	19.17	-
EVA	3.77	-
Nitrogen		
Daily Supply	21.6	-
Cabin Atmosphere	9.45	-
Drinking water	60.48	-
Wash water	34.2	-
Hygiene station	20	0.03
Toiletries	3	0.03

6.3.2 Environmental Control

The crew module utilizes a completely non-regenerative environmental control system. Trade analyses easily show that it is less costly, in terms of mass, to take all the supplies that you need for a six day mission rather than to use regenerative equipment (ie. as in the habitat). Utilizing supplies on a once through basis also is less costly since no additional cost of equipment development is incurred. Basically, the only cost is the mass to the lunar surface and back to the Earth.

Figure 6-28 is a diagram of the crew module's environmental control and waste management system. Table 6-11 contains the total budgets for the system. This system is based on the trade and selection analysis given in Subchapter 7.2 of Volume II and is described in full in the following sections.

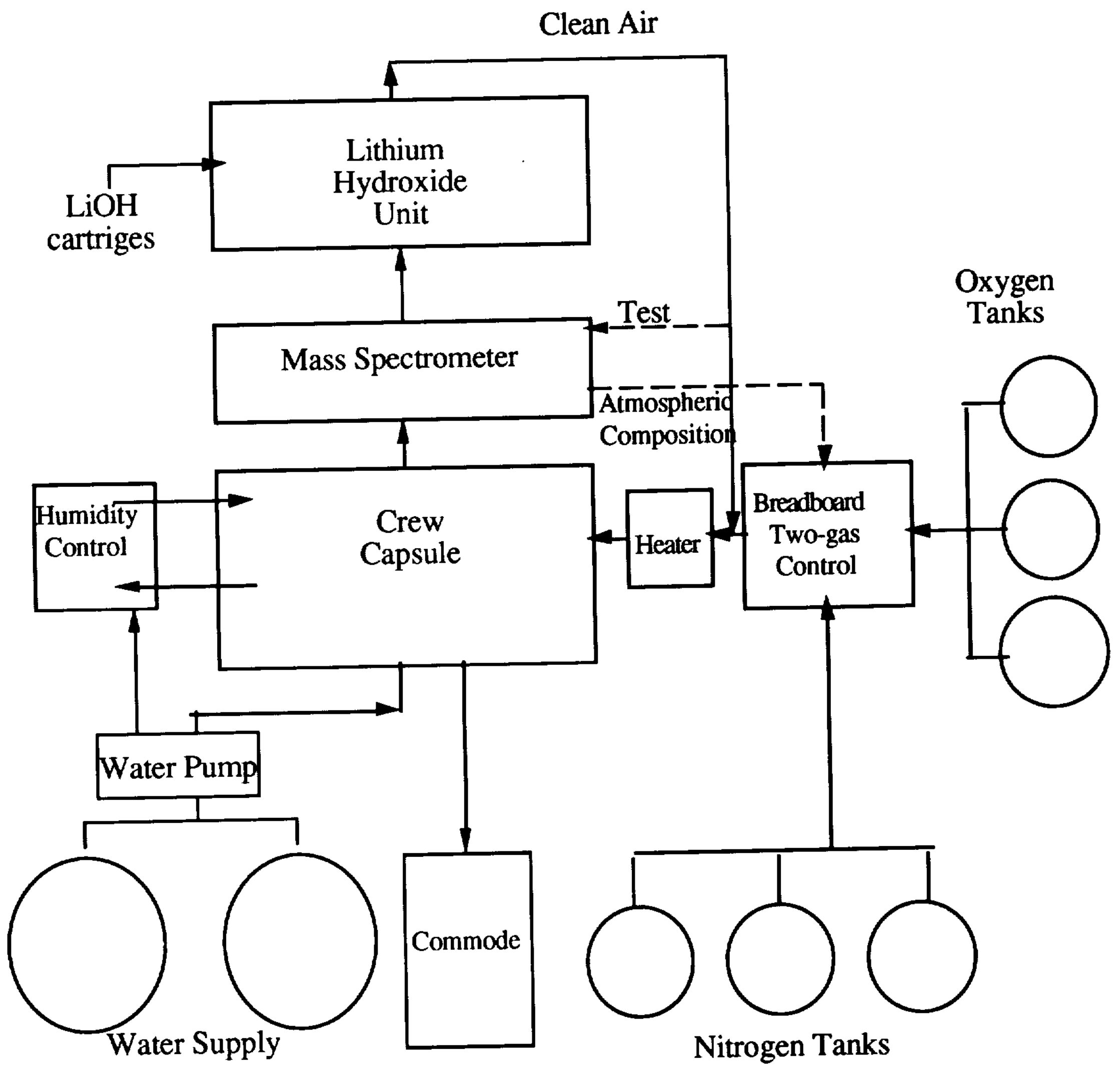


Figure 6-28

Crew Module Environmental Control and Waste Management System

Table 6-11: Crew Module Environmental Control Total Budget

Environmental Control Total	Mass (kg)	Volume (m3)	Power (watts)
	468.94	**5.923**	**2309.8**
Tanks			
Oxygen - three	48	1.59	-
Nitrogen - three	95.7	0.81	-
Water - two	36.34	0.128	-
Waste Management			
Commode	46	0.24	340
Water Management			
Humidity control	55	0.255	725.2
Piping, etc.	14	1	-
Atm. Purification			
LiOH system	20	0.2	20
Thermal Control System	70	0.65	1000.7
Atmosphere Support + Control			
Mass spectrometer	18.2	0.1	100
Breadboard 2-gas control	22.7	0.2	100
Tubing, etc.	18	0.7	-
Fire Suppression and Detection	25	0.05	23.9

The power levels given in Table 6-11 are just the required level for each component. The total given is just a sum of these levels. Subsection 6.3.2.4 contains a power profile for crew systems' part of the crew module.

6.3.2.1 Atmosphere

The general composition of the crew module atmosphere is identical to the atmosphere of the habitat. Section 7.2.1 of Volume II contains the engineering of the atmosphere and the reasons for choosing the following characteristics.

Total Pressure = 0.34 atm

Nominal Partial Pressures =

Oxygen = 0.218 atm

Nitrogen = 0.122 atm

Carbon Dioxide < 0.0102 atm

Water Vapor = 0.0082 atm to 0.0184 atm

Temperature = 17.8° to 27.2 ° C

Mixture (by volume) = 64% oxygen and 36% nitrogen

Carbon dioxide will be removed from the crew module's atmosphere by use of a lithium hydroxide (LiOH) system similar to the Space Shuttle [Joels, 1982 and Pearson, 1971]. The system has lithium hydroxide cartridges which adsorb the carbon dioxide out of the air. The chemical equation (Equation 6-1) is

$$CO_2 + 2LiOH \Longrightarrow Li_2CO_3 + H_2O \qquad (6\text{-}1)$$

The carbon dioxide reacts with the lithium hydroxide to produce lithium carbonate and water vapor which are both waste products. The waste is stored in the cartridges. These cartridges must be replaced every 12 hours during operation. The estimates for the system are 20 kg and 0.2 m^3 (Table 6-11).

6.3.2.2 Water

The crew module does not recycle water by any method (Section 7.2.2 of Volume II). The amount of water needed is provided in Table 6-10. Information on the water tanks and the entire system is provided in Table 6-11.

6.3.2.3 Waste

The Columbiad Crew Module will include one Allied-Signal commode unit for the disposal of human waste, wipes, and potentially other soft disposable items (see Volume II section 7.2.3 for details). In addition to this unit, the capsule will have a location for the storage of garbage materials such as food packaging remmants and used personal hygiene items (dental floss, tissue paper, etc.). This storage area will be supplied with passive air fresheners to eliminate cabin odor. There will be an extensive effort placed on minimizing disposable food packaging. Freeze dried foods will be wrapped in cellophane and eaten on reusable, multi-compartment plastic trays. The cellophane is very compactable and will contribute very little to garbage volume. In addition, beverage powders will be stored in larger storage tubes instead of individual packets to reduce wasteful wrappers. Cups will be reusable.

The waste management equipment will also include a handheld vacuum capable of intaking small liquid and solid spills. Vacuum containment bags will be highly resistant to volatile contents to prevent releakage. Hence, full bags can be placed inside the garbage storage bin for later, more permanent disposal.

The Crew Module Environmental Control System includes an air filter system to reduce atmosphere particulate count to healthy levels. The system also includes a mass spectrometer which will be able to detect trace levels of predetermined expected toxins which will be periodically monitored by crew and mission control.

6.3.2.4 Power

Table 6-11 provided the power levels for the various components of the environmental control system and lighting for the crew module. However, these are just values and do not provide a power profile. All of these systems run continuously except for the commode. The total power that these systems require is 2169.8 watts. The commode runs an average of 14 times per day for the four-person crew. The commode requires 340 watts of power for a duration of 20 seconds each time it is in operation [Shewfelt]. Another aspect of crew systems is bioinstrumentation (Section 6.3.4). In terms of power, the bioinstrumentation requires 100 watts of power for the first two hours of launch or landing. There are four instances of this: launch from the Earth, landing on the Moon, launch from the Moon, and landing on the Earth. Figure 6-29 shows crew systems launch/land daily power profile for the crew module. The commode's power is shown as spikes and the bioinstrumentation as an initial hump. The profile is in terms of a 24-hour period, however, "hour zero" is not necessarily equal to 12:00 a.m.. Figure 6-30 is similar but shows crew systems non-launch/land daily power profile for the crew module.

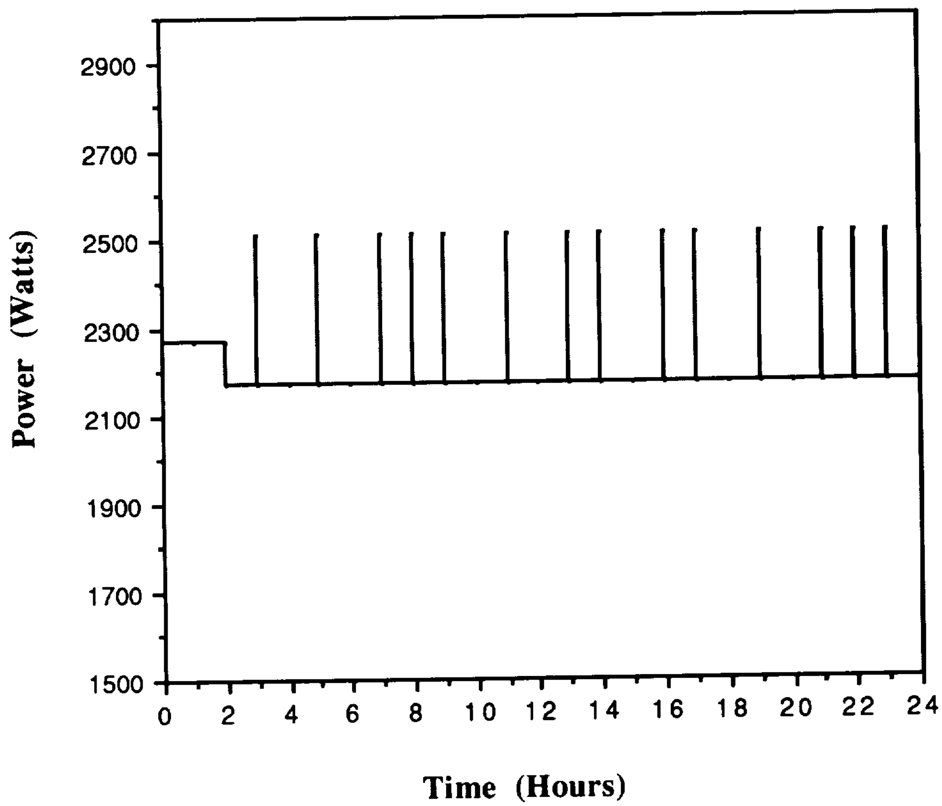


Figure 6-29
Crew Systems Launch/Land Daily Power Profile For Crew Module

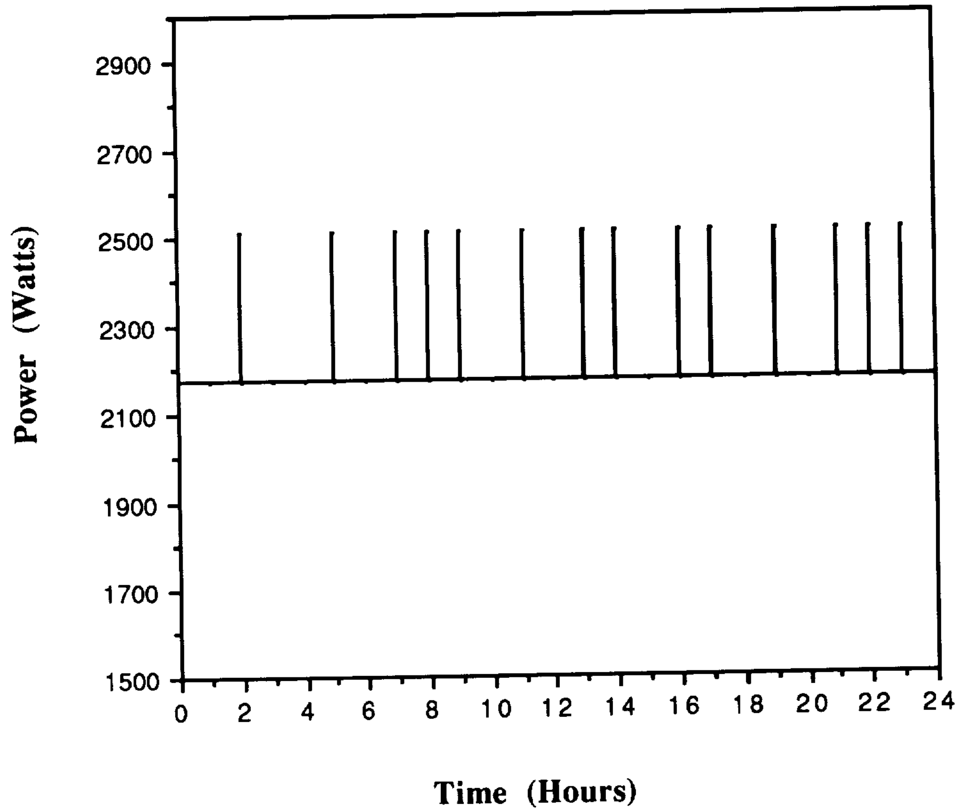


Figure 6-30
Crew Systems Non-Launch/Land Daily Power Profile For Crew Module

6.3.2.5 Fire Suppression and Detection

Fire is a grave danger is space. Possible causes are overheating of electronic equipment and astronaut error. The system implemented in the crew module is very similar to the Space Shuttle's current system. The technology is based on work done at AiResearch [Shewfelt, 1992]. The module contains six smoke detectors, three fire detectors, and three fire extinguishers (one of which is built into the system, two of which are portable). The system mass is 25 kg and volume is 0.05 m^3 (Table 6-11).

6.3.3 Crew Garments and IVA Suit

6.3.3.1 Crew Shirtsleeve and Undergarments

Columbiad crewmembers shall wear a variety of undergarments to remain comfortable both within the IVA pressure suit as well as during non-critical Earth-Moon transit phases when the IVA suits are doffed. This includes a set of Capellene underwear to provide warmth and a layer against outer suit friction irritation. For long term IVA suit wear, particularly during the launch, reentry, and capsule-to-habitat transfer phases, the crew will also don a Fecal Collection System similar to those worn on Apollo missions. This garment is essentially a set of highly absorbent underwear, necessary only in emergencies. During Earth-Moon transit, crewmembers will don Shuttle multi-pocketed pants and flight jackets which provide comfort, warmth, and are highly functional, as well as lightweight tennis shoes.

6.3.3.2 IVA Pressure Suit

The IVA suit chosen is a modified off-the-shelf Air Force full pressure suit, a similar system of which was emplaced on initial Space Shuttle test flights. The Columbiad IVA suit will act as an emergency pressure suit in the event of high altitude ejection, an environment suit in case of water landing, a full pressure suit in the event of cabin depressurization, and finally, the pressure suit by which the crew will transfer back and forth from the capsule to habitat. The Air Force suit, used for high-altitude TR-1, U-2, and SR-71 flights, will be modified by adding medical monitoring equipment, a higher rated pressure bladder, and anti-g protection for the highest launch and reentry loads as well as for possible high load abort options. The suit outer skin will be a high visibility orange color to ensure the astronauts can be found quickly in the event of post-launch or reentry ejection. The helmet will also include an extra protective visor, guarding the wearer's eyes from exposure to high intensity sunlight on the lunar surface.

The IVA suit consists of a torso assembly, a separate helmet, gloves, urine collection system, an anti-g suit, a cooling garment, and the monitoring equipment. The suit has separate breathing and ventilation gas inlets, each with independent plumbing and ducting systems. The suit is supplied through umbilicals from either the under seat mounted life support unit or the portable life support system (PLSS) for the capsule-to-habitat and habitat-to-capsule transfer phases. The regulated oxygen system converts the 5860 kPa oxygen to the 414 to 620 kPa oxygen required for pressure suit and g-suit operation. The suit, therefore continues to be supplied following ejection and initial descent before man-seat separation.

6.3.3.3 IVA Suit Overgarment and PLSS

The garment will most likely be designed according to a relatively recent NASA layup design study [Kosmo,Dawn,1988]. The outer layer will consist of a layer known commercially as Orthofabric which is a layup of Gore-Tex, Nomex, and Kevlar yarns. This provides abrasion resistance as well as an outer thermal layer. Attached to the inside of the Orthofabric is a grid of electrically conductive fibers which aid in dissipating static discharges, and also a chemical contaminants control barrier, probably a thin silicone coating. This is particularly important when in close proximity to propulsion units with potential fuel leaks. Beneath this section will lie multiple layers of alternating aluminized Mylar or Kapton and non-woven Dacron as a low thermal conductive spacer. The layup further includes a radiation attenuating layer of a tungsten-loaded polymeric elastomer and a final layer of micrometeoroid protection in the form of "rip-stop" nylon.

The overgarment will have no pressure capability and so will not be subject to ballooning. This IVA suit overgarment must fit closely over the full pressure suit to allow pass-through of supply umbilical lines and to reduce bunching at the shoulders from PLSS straps. The suit will also include a thermal over-helmet shroud and mittens to allow some level of dexterity while the garment is donned.

Finally, a pair of highly protective overboots similar in design to those used on Apollo suits will complete the protective covering for the IVA suits. These are specially designed to drastically reduce the possibility of severe abrasion and puncturing which could lead to suit decompression. The Apollo [Kosmo, 1988] overboots were constructed primarily of stainless steel woven fabric (Chromel-R) with the tongue area of the boot made from teflon-coated Beta cloth (fiberglass). Inside of the Chromel-R fabric consisted of two layers of Kapton and five layers of aluminized Mylar film separated by four layers of non-woven Dacron and a liner of teflon-coated Beta cloth. The innermost sole was finally covered with two layers of Nomex felt to provide an extra thermal barrier. A rib structure on the soles provided increased thermal insulation qualities, to provide lateral rigidity, and to provide traction on the lunar surface. The very bottom of the sole was covered with a layer of silicone rubber to add grip to the boots. The Columbiad boots will undoubtedly be very close in design to those of the Apollo missions.

The IVA Transfer PLSS unit is a miniaturized version of PLSS units used on Apollo, with several modifications. It will be in backpack form with straps and waistbelt long enough to go over the IVA suit overgarment. All supplies from the PLSS enter the IVA suit via two umbilicals. The system supplies the spacesuit with a 100% oxygen supply at 0.34 atm. It will provide only 4 hours of nominal oxygen supply as well as 1/2 hour worth of emergency oxygen. This capacity should be sufficient to allow four astronauts to complete the habitat setup including cabin pressurization and thermal control stabilization necessary to eliminate PLSS dependence. Unlike the EVA PLSS, exhaled air is processed through a lithium hydroxide cartridge system instead of the larger molecular sieve system. As in the Crew Module, the cartridges must be replaced as they turn into lithium carbonate. In addition, the PLSS is integrated with a liquid cooling garment to provide a comfortable temperature for the working astronaut. The working fluid is transported to the pack, and sublimated, releasing the heat to a radiator and outer space. Finally, the backpack includes a battery which supplies all mechanical systems such as pumps with power. The system is compatible with the PLSS recharge system included on the lunar habitat (see section 8.1.3.5).

6.3.4 Bioinstrumentation

In order to monitor and maintain crew health, a multichannel electrocardiogram, MK-I Exerciser and First Aid Kit will be supplied on the crew capsule.

Electrocardiogram. Data on heart electrical activity of each crew member will be obtained via electrodes worn under flight clothing. ECG monitoring will be necessary only when rapid variations in g-loading induce stress on the human cardiovascular system. In program Columbiad, continuous ECG monitoring of crew will occur during:

- earth and lunar launch
- earth and lunar re-entry
- earth and lunar landing
- the first hour after launch and re-entry

(see Volume II: Section 7.5.1)

MK-I Exerciser. Almost immediately after lunar landing, astronauts must perform strenuous EVA activities to set up the lunar habitat. Decreases in muscle strength needed during the weightlessness of earth to moon transit necessitates the use of the MK-I exerciser on the crew capsule. This isokinetic device has been shown to be effective in maintaining arm strength by providing a resistive force which counters the force applied by the user. The low volume and mass of the MK-I makes this item suitable for crew capsule use. (see Volume II: Section 7.5.2)

First Aid Kit. Minor injuries and inflight illness will be treated with equipment and pharmaceuticals included in a first aid kit. Medications will only be supplied for the short duration of capsule habitation. (see Volume II: Section 7.5.3)

Characteristics of these three items are provided in Table 6-12.

Table 6-12: Crew Capsule Bioinstrumentation

Parameter	Multichannel ECG	MK-I Exerciser	First Aid Kit
Number Supplied	1	2	1
Dimensions Height (m) Depth (m) Length (m)	 0.14 0.39 0.46	 0.20 0.52 0.20	 0.39 0.13 0.26
Volume (m^3)	0.03	0.02	0.01
Mass Per Item (kg)	9.00	5.49	3.00
Power Per Item (W)	100	-----	-----
Cost Per Item ($)	5000.00	75.00	75.00
Supplier	Siemens-Burdick	NASA	Zee Medical

6.3.5 Other Equipment

Subchapter 7.7 in Volume II provided the additional equipment and selection for the crew module. Table 6-13 provides the budgets for this additional equipment.

Table 6-13: Additional Crew System Equipment for The Crew Module

	Mass (kg)	Volume (m3)	Power (watts)
Other Equipment TOTAL	**33.5**	**0.12**	**200**
Lighting	4	0.01	200
Tools, cleaning equipment	29.5	0.11	-

6.4 Guidance, Navigation, and Control Systems for the CM

6.4.1 Installation of IMU

The IMU is that discussed in Section 5.2.3 of Volume II. It must be installed in the command module so that the gyros and accelerometers are aligned with the spacecraft coordinates. The location of the IMU in the spacecraft is not critical, but if it is located away from the center of mass, rotations that occur in translational manuvers must be subtracted from the input data. In this case, the location of the IMU with respect to the center of mass is very important.

6.4.2 Storage of Ephemeris

The ephemeris, which consists of a star catalog and the desired state vector versus time, must be stored on the CM, either onboard or on the ground. The ephemeris is used to supply the star trackers with information on what stars are visible when the spacecraft is in a certain position. The ephemeris is orbit dependent, and requires that the actual trajectory follows the planned trajectory very closely.

6.5 Structures

The following is a highlight of the crew capsule configuration and structural design the is described in detail in Volume II - section 2.2.4.

6.5.1. Sizing and Configuration

The final geometrical configuration of the crew capsule is shown in figures 6–31 and 6–32. Table 6–14 summarizes the dimensions of the biconic crew capsule with a $45m^3$ inside volume.

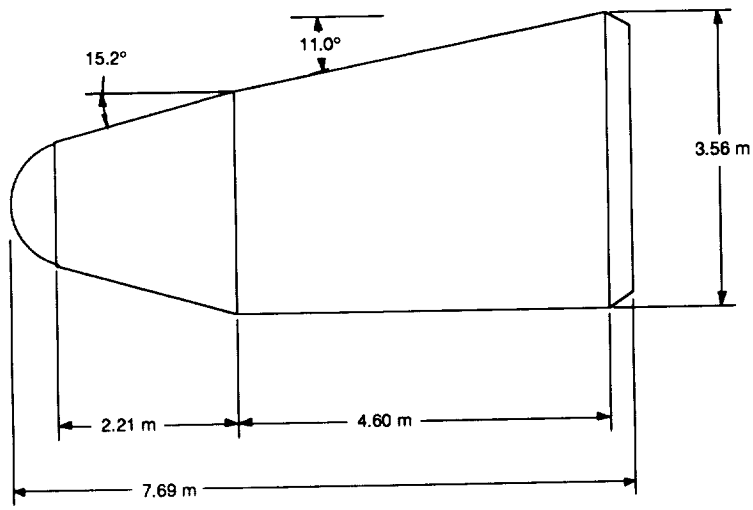


Figure 6-31
Crew Capsule - Side View - Dimensioned

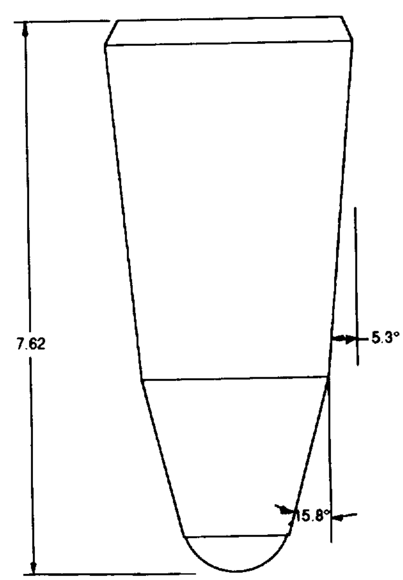


Figure 6-32
Crew Capsule - Top View - Dimensioned

Table 6-14: Summary of Geometric Configuration

Geometric Configuration	
Maximum Diameter	3.55
Total Area	86.89
Total Volume	46.22
Extended Width	7.19
Total Length	7.85

6.5.2. Structural Design and Loading

By using the structural design process described in Volume II section 2.1.3, the crew capsule was designed using a solid monocoque structure, and the conversion factors were used for determining the mass of a semi-monocoque design with the same structural state.

For the structural design we need to know the maximum loads experienced by the vehicle. Since during launch it is enclosed in an aerodynamic fairing, it experiences only the 3.5g's acceleration and no aerodynamic loads. However, during reentry it experiences 1g acceleration plus the aerodynamic pressure. According to Figure 6–33 the maximum dynamic pressure is about 8,000 Pa. This, in addition to a 3.5g acceleration load experienced during launch, are the design loads for the Crew Capsule.

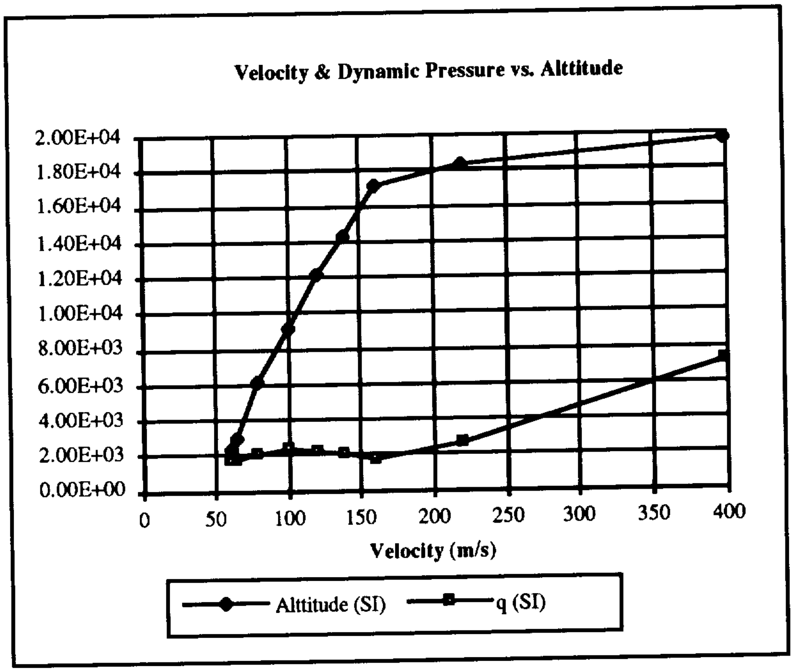


Figure 6-33
Velocity and Pressure Trajectories

After several design iterations, a 1.5cm solid aluminum structure resulted in satisfactory stresses and deflections. This configuration is very capable of carrying the applied loads. It is under-stressed to allow for increases in the un-modeled internal mass of the crew capsule.

The 1.5cm aluminum skin thickness produces about a 3600kg total structure weight. This is shown in Table 6-15.

Table 6-15: Skin Thickness and Weight for a Solid Monocoque Aluminum Structure

Solid Aluminum Weight	
Skin Thickness	0.015
Aluminum Total Weight	3610.23

As was shown in section 2.1.3, the weight of a semi-monocoque structure that induces the same stresses and deflections as a solid monocoque one, is about 50% of the weight and about 2.5x as thick of the monocoque structure. If our Crew Capsule weight based on a solid aluminum structure is 3600kg, our semi-monocoque structure will weigh approximately 1800kg. With the weight of the wings, and wing deployment system, the final structural weight is 2000kg.

6.6 Thermal Control Systems for the CM

6.6.1 Heat Shield Design

The crew capsule uses three types of radiative insulation. They are summarized in Table 6-16. They are listed in order of descending tolerance to heat flux. Also listed in Table 6–16 is the required thickness.

Table 6–16: Insulation Materials

Material	Abbreviation	Density	Thickness
Lockheed Insulation	LI-2200	353	0.063
Fibrous Refractory Composite Insulation	FRCI	388	0.058
Tailorable Advanced Blanket Insulation	TABI	258	0.0127

Table 6–17 summarizes the insulation covering of the crew capsule. The heat shield is broken down by area, and is listed according to the amount of insulation covering each section. For instance the entire nose needs to be covered with LI-2200 since it experiences the maximum heating and LI-2200 is the only insulation that can withstand that environment.

Table 6-17: Calculation of Heat Shield Coverage and Weights

Location	Area	Material	Weight
Nose Surface	3.3277	LI-2200	74.00
Wing % Heat Shield		0.25	
Wing Leading Edge	1.5090	LI-2200	33.56
Wing Surface	4.5269	FRCI	101.87
Frontal Cone % Heat Shield		0.25	
Frontal Cone Lower Surface	3.7120	FRCI	83.54
Frontal Cone Upper Surface	11.1361	TABI	36.49
Main Cone % Heat Shield		0.2	
Main Cone Lower Surface	8.9725	FRCI	201.92
Main Cone Upper Surface	35.8901	TABI	117.60
Boat Tail % Heat Shield		0.2	
Boat Tail Lower Surface	2.3557	FRCI	53.01
Boat Tail Upper Surface	9.4229	TABI	30.87
Removable Material Weight			412.47
Support Structure Weight			254.40
Attachment Mechanism Weight			25.44
Removable Shield Weight			692.31
Weight Remaining w/Vehicle			320.39
Total Weight of Removable			**1012.70**
Total Integrated Shield Weight			**732.86**

The heat shield will be permanently attached to the crew capsule. Therefore, the total weight of the heat shield is 732kg.

6.6.2 Heat Pipe System in the CM

6.6.3 Thermal Insulation in the CM

While in outer space and on the lunar surface, it will be necessary to provide additional thermal control for the Command Module. This will in part be provided by the standard reflective coating on the outside of the craft; however, insulation will also be required in order to maintain the interior of the crew capsule at habitable temperatures. It is also worth noting that this insulation must keep too much heat from escaping during the 14 days that the capsule is on the dark side of the moon, as well as protecting it from solar radiation during the rest of the mission.

Double-quilted fibreglass cloth, with a density of 83 kg/m3, will be used for this purpose. This blanket should be 6 centimeters thick; it will cover 26.08 square meters of area, occupying a total volume of 1.3 cubic meters and having a mass of 110 kilograms. This should limit the interior capsule temperatures to a range between 17.8 and 22.7 degrees Celsius.

Although a nylon screen insulation could serve a similar purpose while weighing slightly less and occupying far less space, fibreglass cloth has been selected on the basis of its flexibility, which allows it to conform more smoothly to the interior contours of the Command Module.

6.7 Power Systems in the CM

The CM has been designed for a seven day mission, allowing 3.5 days for the journey to the moon and 3.5 days for the return to Earth. This mission plan allow for a consumables design margin of 0.6 days for the longest mission time, a launch to a lunar pole and back.

For most of its trip, the CM will be powered by cells on board the ERM, via the connection of a power bus. However, for the last two hours of its journey, during the Earth reentry, the CM will be powered by an onboard power cell. During this last two hours: the following power will be needed.

- Maximum applied wattage of 4877 W.
- Total work of 9.49 KW*hr.

This power and work is distributed among Crew Systems, CCC, GNC, Status, onboard RCS, and Structures, as shown in the Figure 6-34.

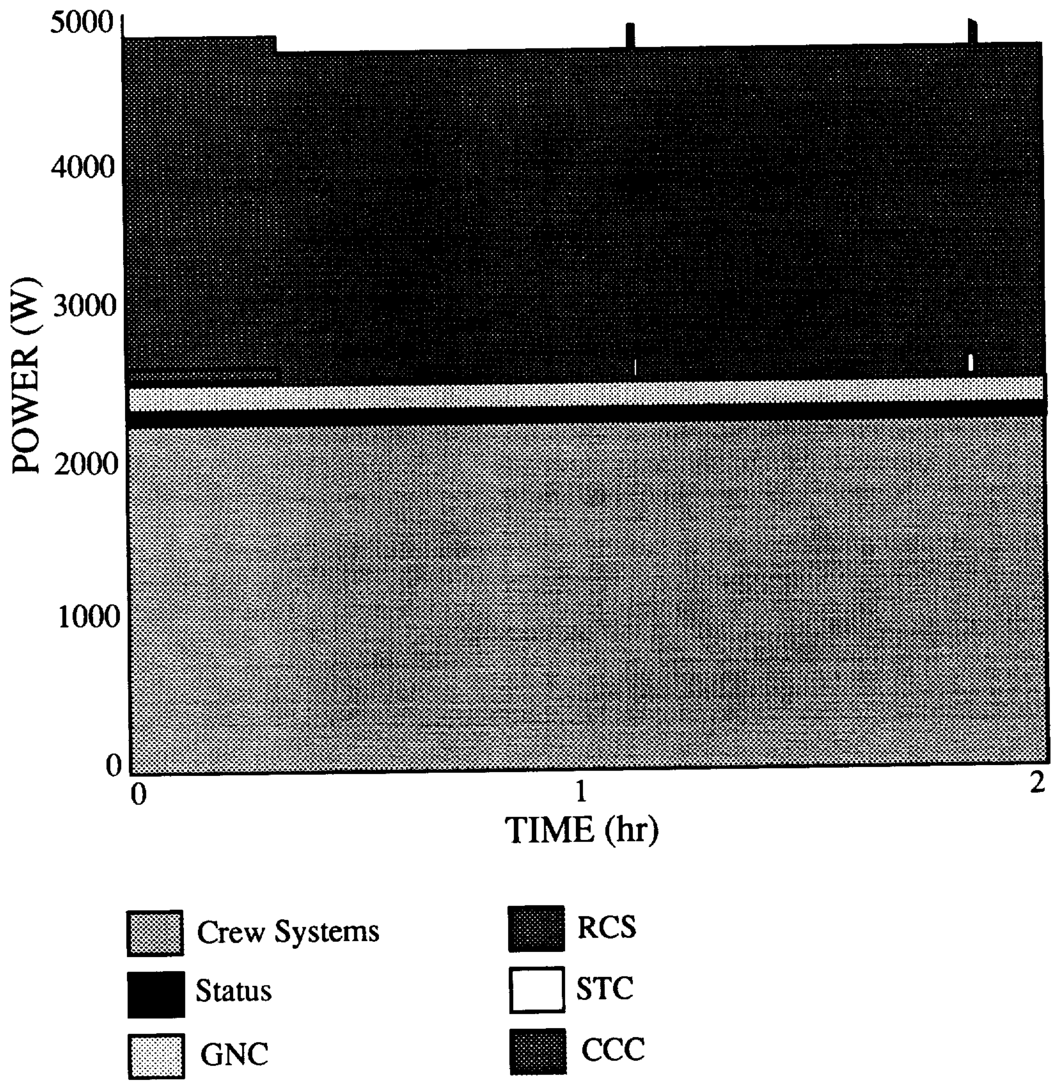


Figure 6-34
Power Curve for Onboard Fuel Cell on CM

Crew Systems requires 2170 W of power at constant application to run all atmospheric and thermal control systems. An additional 340W of power, applied for 20 seconds is needed to run the commode. Crew systems has budgeted 14 uses of the commode per mission day, but no uses of the commode are allowed during the last 2 hours of approach and reentry. For the first two hours after Earth launch, and during the last two hours of Earth approach and reentry, however, crew systems needs an addition 100 W for biomedical monitoring of the crew members' life signs.

Command, control, and communications needs a constant source of 2182 W to run all computers, modems, demodulators, receivers, transmitters, and data storage. These systems will continue to run during the last two hours of the mission, continuosly updating the information given to the astronauts.

GNC also requires continuous power usage through the entire mission. All onboard sensors, monitors, navigation aids, and control devices require a total of 175 W.

The status monitoring both in the form of onboard information and telemetry information beamed back to mission control requires a continuous 100W of power at all times.

Structures requires no power throughout the mission, expect for four bursts of power during reentry. Two bursts of 150W, applied for 10 seconds will be required to deploy the wings and lower the landing gears. Two small 100W spikes of 1 second are required to deploy both the drogue chutes and the parafoil.

The CM will rely on the ERM's, the LBM's, and the PTLI's RCS systems through all but the last two hours of the mission. At this time the CM is conservatively budgeted for 20 minutes of continuous operation of its RD-4 engines, firing in bursts of two of its eight engines at one time, requiring a maximum of 100W.

The onboard power system to supply this power for the last two hours weighs 21.6 kg and has the dimensions of 30cm x 30cm x 46 cm. This system is located behind the LOX tanks at the rear center of the CM.

6.8 Command Module Mission Profile

6.8.1 Launch Pad

6.8.1.1 Pre-engine ignition

The Crew Module is the "nerve center" of Project Columbiad. It is the vehicle in which the four crew members will be transported from the Earth to the lunar habitat and back. In the effort to minimize cost, complexity, and weight, the CM must be a concise and complete living environment in addition to the systems control center for the entire launch system. For this reason, there are thousands of interfaces and systems which must maintain a high level of performance, reliability, and safety. Once the launch vehicle has been transported to the launch pad, the Crew Module is a vital part of the spacecraft and goes through many pre-ignition checks. As the launch vehicle sits on the pad, the astronauts sit with their backs to the ground in their soft IVA spacesuits. Ground control performs multiple systems checks, complete with everything from structural to status testing. They are hooked up to ECGs from before the launch until rendezvous with the PTLI stage (see Section 6.3 for more information) which monitor the crew's life characteristics for the first few hours of flight. In the case of an abort, the crew will evacuate the area with a slidewire egress system based on the Space Shuttle. After exiting the CM and crossing the egress catwalk, multiple slidewire baskets bring them to safety.

6.8.1.2 Post-engine ignition

Immediately following ignition, the launch vehicle will actually remain on the pad for a few seconds, during which an abort situation requiring ejection is highly risky. This is due to the fact that a reasonably large explosion may injure or kill the astronauts while they are descending in their parachutes.

6.8.2 Earth Surface to Orbit

Next, a fully automated sequence of ignitions ensue as the astronauts sit back and enjoy the ride into Earth orbit. The first abort mode stage is the ejection seats which will be initiated by the Mission Commander or by the Range Safety Officer from ground control. This abort choice can only be utilized until the spacecraft reaches an equivalent airspeed of 308 m/s and occurs through two hatches, one above each pair of seats. Once far enough away from the vehicle, the crew then undergoes a completely automatic procedure with the parachutes .

In terms of abort options, there is currently a short window after ejection into the Earth's atmosphere is possible during which there is no abort mode. Since the SRB's are pushing the entire structure at a very rapid climb rate, the Earth Return Module cannot provide enough thrust to propel away from the boosters in case of emergency. Once the SRB's burn out, then the abort mode of choice is to fire the ERM rockets and then follow one of numerous paths to a safe recovery zone. These include Trans-Atlantic Abort (TAL), Abort-Once-Around (AOA), and Abort-to-Orbit (ATO) and are detailed in the Abort section of the Propulsion chapter. If all goes well and abort is avoided, then the astronauts will maintain a course into LEO where they will once again test systems and check to make sure all is working properly for the trans-lunar injection.

6.8.3 Trans-Lunar Injection

The trip between Earth and the Moon takes approximately three days, during which the astronauts have a fairly easy job. They will be confined to a very small area (total of 1.8 m per crew member) and thus cannot perform any large experiments or be expected to perform any significant tasks. They will eat, sleep eight hours a day, go to the bathroom as necessary, and maintain status, all the while trying to fight boredom and other psychological problems. The only major procedure which the crew must perform during this phase is the changing of the lithium hydroxide cartridges every 12 hours. The addition of a small window to the CM will hopefully alleviate some of the harmful mental effects of living in a box with three other people for three days.

During the three days, they will be allowed to remove their soft suit helmets but will most likely remain in their body suits because of safety issues and space constraints. Abort mode choices in this stage all involved the firing of the ERM, so that the CM will merely follow a specified path to Earth.

6.8.4 Lunar Descent and Landing

Since the mission is direct flight, there will be no transit to lunar orbit, so near the end of PTLI burn the astronauts will return to the chairs in full IVA suits. Much like launch from Earth, they will have their backs facing down to prevent g-loading problems. The Lunar Braking Module will slow the capsule down near the surface and after this stage is jettisoned, the ERM will be used to hover to the surface. The lunar descent maneuvers are completely autonomous while the ERM hovering and landing are to a certain extent manually handled. With the use of two sets of redundant cameras, the Mission Commander will use a stick control to choose the ERM/CM's landing site and it will come to a soft rest on the lunar surface. The stick will only tell the computers on board where the MC desires to go and will then automatically provide all the RCS reactions accordingly. This type of automation will not allow the spacecraft to go unstable or perform a maneuver which would go beyond acceptable limits. This computer redundant system is similar to many aircraft and spacecraft today in which manual control is used with automatic procedures and overrides.

6.8.5 Lunar Stay

6.8.5.1 First Three Hours

After the spacecraft lands and the engines are turned off, the astronauts must run through a series of system checks for both the CM and the lunar habitat. Once all is deemed safe and operational, two of the astronauts will add an EVA overgarment and boots over their IVA, attach a Portable Life Support System to their suits, and then prepare for a journey to the habitat. The hatch will hydraulically open and a pulley system will be pushed out so that crew members can easily descend to the surface. See Section 6.2.6 for more details about the egress system design. The first two astronauts will descend and walk to the habitat to put on their EVA hardsuits. Meanwhile, the remaining two crew members in the CM will close the hatch and await the return of the first two who will bring them their EVA hard suits, located in the habitat. Abort during this time can be accomplished within minutes because all systems are still at full power and only the ERM needs to be fired to leave the surface.

6.8.5.2 Remainder of Stay

Once in the hard suits, the four astronauts will work in teams to perform many initial checks of the habitat and other necessary procedures. One of these will be to connect a long umbilical from the habitat to the CM, so that during the 28 days, the CM can be maintained using power from the habitat and the CM's systems can be monitored from the lunar surface. Of course, the CM will have enough power of its own to run for the month-long stay, but this will only be used as a backup mode in case of umbilical or habitat failure. After all the astronauts are safely in the habitat, the pulley system will be retracted and the hatch will be closed by way of remote control. Systems on board the CM are powered down and the environment is repressurized with nitrogen gas for the duration of the stay. This will be achieved by turning the oxygen valve off and leaving the nitrogen valve on, creating an environment that will help preserve the systems on board the CM for the remainder of the stay. Through the umbilical, any of the systems can be monitored and adjusted. When the astronauts are ready to leave the Moon, they will begin to power up the systems 24 hours in advance and exchange the nitrogen with a habitable atmosphere. For the purpose of powering up the systems, a 4-6 hour window is required for an abort to lunar orbit, a choice available at any time during the lunar stay.

6.8.6 Lunar Ascent

Before entering the CM, the astronauts get into their IVA softsuits with anti-abrasive shell and return to the landing site. With the help of the pulley system, the four will return safely to the CM and perform numerous system checks. Prior to ignition, the astronauts remove the anti-abrasive layer, seat themselves with backs facing down, and prepare for liftoff. Procedure is similar to the launch pad process, except there are significantly fewer systems to worry about. The ERM fires and the spacecraft is lifted away from the Moon.

6.8.7 Earth Return Stage

Much like the PTLI stage, the astronauts may eat, sleep, go to the bathroom, remove their helmets and do anything to pass the time. They must change the lithium hydroxide cartridges and perform any necessary repairs. Naturally, they will still have to monitor their course to make sure the computer navigation systems are performing their job.

6.8.8 Descent into the Earth's Atmosphere

6.8.8.1 Ballistic Trajectory

See Section 6.2.7 for specific information regarding the CM's stages and maneuvers during reentry.

6.8.8.2 Drogue/Parafoil Deceleration

During the parachute stages, the crew will only be monitoring the systems to make sure all is well. Commands involving the drogue chute release and parafoil disreefing are not the responsibility of the crew and are completely autonomous. Using the joysticks, the crew will be able to control the parafoil so that landing is somewhat crew-controlled. Similar to the CM/ERM landing on the lunar surface, the system will not allow the parafoil to go unstable and will actually perform the maneuvers while the crew only tells is where it wants to go. See Section 6.2.7.4 and 6.2.7.5 for details about the drogue chute and parafoil.

6.8.8.3 Landing

The CM will hit the Earth at a vertical velocity of 1.5 m/s and a horizontal velocity of 20 m/s under a crew-guided system. With Edwards Air Force Base in California as the primary landing site, both White Sands, New Mexico and Hawaii are being investigated as secondary landing sites. Once touchdown occurs, the parafoil is released by pyrotechnic cutters and the landing gear brakes decelerate the CM to a stop. The crew have been safely returned to Earth.

6.9 Crew Member Roles

The crew will consist of four members with the following designations: Commander, Co-pilot, Medical Specialist, and Maintenance Specialist. These designations are not exclusive; however, they supersede any auxiliary roles the crewmembers may play with regard to other aspects of the mission in the case of crisis.

6.9.1 Chain of Command

The Commander is in local control of the mission, followed by the Co-pilot, Medical Specialist, and Maintenance Specialist. All cogent crew members are expected to provide information to the crewmember-in-charge; however, that individual is ultimately responsible for making the local and emergency decisions regarding the mission. The chain of command will be followed in the case of incapacitating injuries or death of the commander.

6.9.1.1 The Commander

The Commander is in local control of the mission. The Commander will make all final decisions regarding mission success not requiring approval of Ground Control, or in the absence of contact with ground control. The Commander is the primary pilot for the mission, and will perform these duties if able. The Commander will be the last to leave the CM on the lunar surface, first to return upon departure.

6.9.1.2 The Co-pilot

The Co-pilot is the second-in-command and as such must be able to land the CM on the Moon and Earth in the absence of the Commander.

6.9.1.3 The Medical Specialist

The Medical Specialist must have expertise in the treatment of trauma in order to reduce the risk of crewmember incapacitation or death in case of injury.

6.9.1.4 The Maintenance Specialist

The Maintenance Specialist must have expertise in the maintenance of the vital systems of the Crew Module in order to increase the fault tolerance of the spacecraft and assist in crisis management.

6.9.2 Duty Shifts

6.9.2.1 Regular (Transit) Duty Shifts

During transit between the Earth and the Moon, the crew will be split into two duty pairs: the Commander and the Maintenance Specialist, and the Co-pilot and the Medical Specialist. The on-duty crew may not sleep, and are responsible for dealing with any problems that may occur during their shift. Each duty shift will be 12 hours long. Crewmembers on duty will sit in the forward most seats at the Instrument and Control Panel.

6.9.2.2 Maneuver Shifts

During shifts which contain maneuvers (ie. launch, landing, or docking) the forward most seats will be occupied by the Commander and Co-pilot, and all crewmembers will be on duty.

6.9.3 Mutual Training

All crewmembers will receive moderate training in all disciplines, including piloting of the spacecraft. However, for the purposes of each mission, each crewmember will have a specific responsibility, and will have a high level of expertise in that area.

7. Payload Landing Module

7.1 Stage Requirements and Operations

7.1.1 Requirements

The Payload Landing Module (PLM) is responsible for landing and deploying the precursor payload on the Moon.

The PLM activates 600m above the lunar surface. First, the PLM distances itself from the LBM. Next, the PLM controls the mission's descent to the surface. Once near the lunar surface, the PLM gently lands the payload in a vertical orientation. Once the mission has landed, the PLM topples itself, and sets itself down in a horizontal orientation. Finally, subsystems are deployed to initiate unpiloted lunar operations.

7.1.2 Budgets

The PLM stage has weight, volume, power, and propellant constraints. Each component requires a portion form each of these budgets. Table 7-1 shows the mass breakdown in the PLM stage.

Table 7-1: Mass Breakdown for PLM

Component	Mass(kg)
Structure	
Wall Structure	2460
Rocket Truss	227
Struts and Bracing	500
Propellant Insulation	726
Propellant Tanks	362
Joints, fittings, trusses to hold components	350
Gang Plank	160
Habitat Support Legs	292
Main Propulsion System	
3 RL10A-4 Main Engines	504
LOX	8826
LH	1276

Helium Pressurization	211
Actuators for gimballing	60
Valves, piping, etc.	50
RCS Propulsion System	
16 R4-D RCS Engines	58
MMH & Tanks	202
N_2O_4 & Tanks	327
Helium Pressurization	8
Valves, piping, etc.	88
Deployment Engine System	
3 Star 48/TE-M-236 Solid Rockets	48
2 XLR-132 Deployment Engines	108
MMH & Tank	28
N_2O_4 & Tank	46
Helium Pressurization	1
Valves, piping, etc.	20
Subsystems	
GNC Suite	61
C^3 Antennae	41
Fuel Cells	306
SLURPP Components (reliquifaction,converters,etc.)	592
Auxiliary PV Arrays	50
Water Bladder	20
Sensors	20
Data and Power Lines	10
Cargo	
SLURPP PV Arrays	274
SLURPP Array Structure and Motors	200
Lunar Rover	950
Regolith Bagger	1500
Regolith Bag Conveyor Belt	1000
Regolith Support Structure	3038
Habitat	9429
Total	**33508+**

Table 7-2 shows a volume breakdown. The Cargo Bay of the PLM is 2.5m long and 3m in diameter, allowing for a maximum of 71m3 of cargo carried in the PLM. This space is alloted to the lunar rover, regolith bagger, SLURPP PV arrays, regolith conveyor, and miscellaneous cargo. Additional volume is available inside the habitat if the cargo exceeds this limit.

Table 7-2: Volume Allotment for PLM

Component	Volume (m^3)
Main Propulsion System	
3 RL10A-4 Main Engines (2.29m long* 1.2m diam.)	8.27
LOX	7.74
LH	17.92
LOX Insulation	6.73
LH Insulation	10.35
Helium Pressurization	0.15
Valves, piping, etc.	0.5
RCS Propulsion System	
12 R4-D RCS Engines (0.56m long*0.28m diam.)	0.06
N_2O_4	0.13
Valves, piping, etc.	0.5
Deployment Engine System	
3 Star 48/TE-M-236 Solid Rockets(0.324m long)	0.1
2 XLR-132 Deployment Engines(1.2m long*0.6m diam.)	0.68
MMH Tank	0.32
N_2O_4 Tank	0.03
Valves, piping, etc.	0.2
Subsystems	
GNC Suite	0.2
C^3 Antennae(2 unmbella dishes*3m diam.)	0.2
Fuel Cells	XXX
SLURPP Components (reliquifaction,converters,etc.)	XXX
Auxiliary PV Arrays	1.5
Water Bladder	6.47

Sensors	0.25
Data and Power Lines	0.15
Cargo	
SLURPP PV Arrays	14.94
Lunar Rover	15.6
Regolith Bagger	20
Regolith Conveyor	12
Total	**126+**
PLM Maximum	**252**

Table 7-3 shows power requirements the PLM must provide for. The engine starts and the landing radar only draw power for a short period. The only significant power consumer is the computers aboard the habitat which must always be running.

Table 7-3: Power Allocation for LBM

Component	Power	
RL10A-4 Engine Startup/Valves/Shutdown	1500W	3*(25V, 20A)
R4-D Engine Startup/Valves/Shutdown	150W	6*(25V, 1A)
Navigation Equipment	37W	
Landing Altimeter Radar	100W	
Commmunications Antennae	93W	
Computer Equipment in Habitat	2129W	
Sensors	20W	
Power Required (continuous)	**2279W**	
Power Required (peak)	**3929W**	

The propellant tanks in the PLM contain both fuel for propulsive purposes and power purposes. The two systems share the same storage facilities in the PLM. Table 7-4 summarizes the breakdown of the cryogenic propellant carried by the PLM.

Table 7-4: Cryogenic Propellant Allocation in PLM

Mission	Mass LH (kg)	Mass LOX(kg)
Power for LBM, PLM, Habitat during flight	6	46
Hovering and Landing (ΔV = 500 m/s)	551	3031
SLURPP Energy Storage for Lunar Night	719	5749
Total	**1276**	**8826**

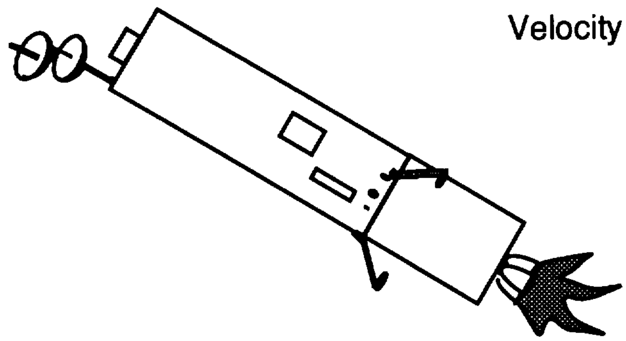
7.1.3 Mission Profile

7.1.3.1 Staging from LBM

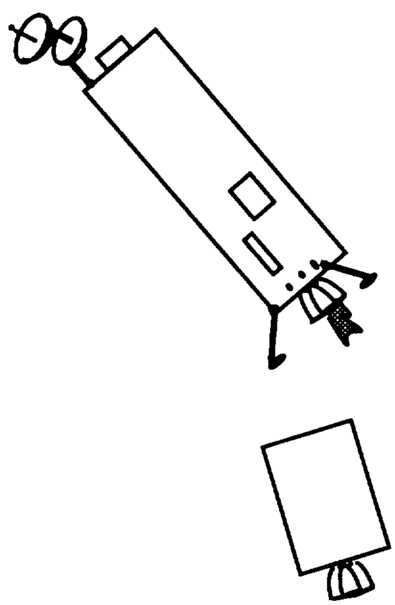
The LBM positions the precursor lander 600m from the lunar surface. At this point, explosive bolts fire between the stages and the LBM is discarded. Simultaneously, the PLM main engines start, and thrust to distance the PLM from the LBM stage. The PLM follows a slanted trajectory such that it lands 1km downrange of staging. This condition insures that the LBM does not careen into the lunar habitat.

7.1.3.2 Hover and Landing

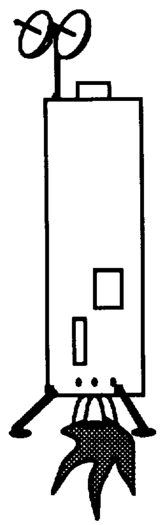
The PLM starts by thrusting to distance itself from the discarded LBM so that it does not interfere with the landing site or sequence. Then the PLM allows the precursor lander to slowly progress toward the surface. Once the flight has neared the surface, the PLM provides enough thrust to hover until an appropriate landing site is confirmed. Ideally, a landing site is already chosen and maneuvered to during descent. However, final corrections still may be needed so the hover option and sufficient fuel is provided. Once, the flight is above the targeted landing site, the PLM lowers the flight the final distance and allows the flight to touch down as gently as possible. Throughout the flight, the PLM is responsible for keeping the flight in a vertical position since this attitude allows for proper descent braking and landing on the landing struts located at the base of the vehicle. A rough sketch of the landing sequence is shown in Figure 7-1 on the next two pages.



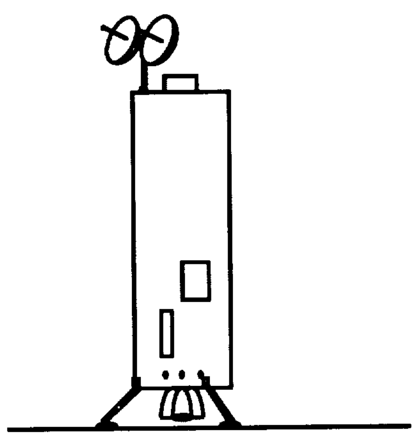
1) Lunar Braking Module (LBM) burns to Brake Lunar Orbital Velocity of Vehicle



2) The Precursor stages from the LBM above the Lunar surface. The Payload Landing Module (PLM) ignites to finish braking and landing.



3) The PLM orients the Precursor parallel to the lunar surface. The PLM allows the Precursor to slowly descend the remaining distance from the Moon.



4) A final landing site is selected, the Precursor hovers above it, and touches down softly on the Moon.

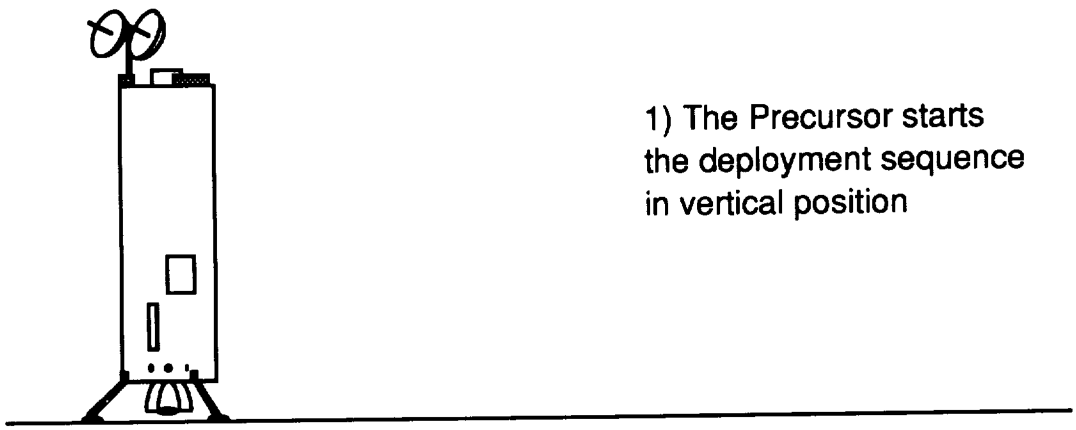
Figure 7-1
Precursor Landing Sequence

7.1.3.3 Tip-over

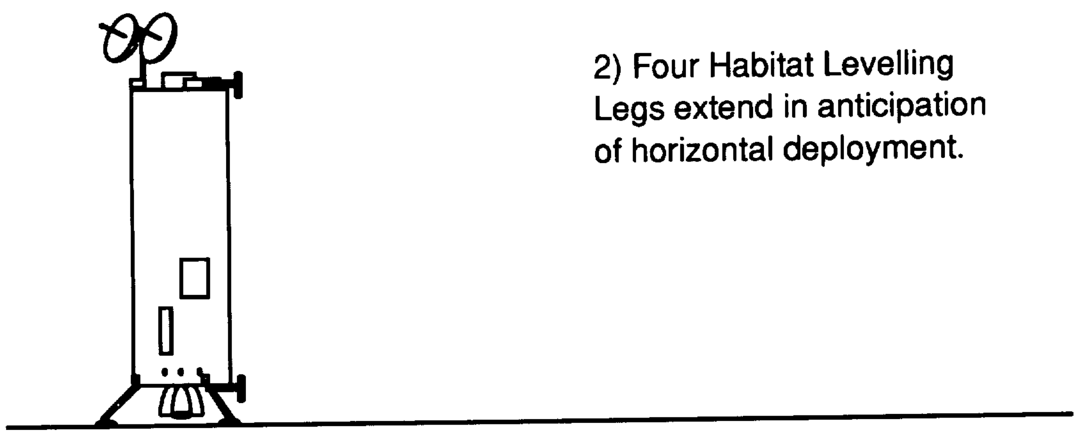
Once the precursor lander has touched down upon the lunar surface, the PLM is responsible for positioning the precursor payload and setting up the habitat as much as possible before personnel arrive. The power, thermal, biological, and structural systems must be checked out as operational before the staffed mission will launch. The bulk of this condition is that the habitat must be reoriented from vertical to horizontal so that the astronauts can enter the habitat and don hardsuits before setup work is performed. The mechanism by which the precursor mission is reoriented is: 1) An impulse from solid rockets to tip the precursor lander to an unstable position. 2) A controlled angular reorientation to horizontal by liquid rockets.

Three tailored Star 48/TE-M-236 are the solid impulse rockets. The engines burn for 6.5 seconds, allowing the center of mass of the precursor lander to cross the stability region 6.9 seconds after the rockets were started. The angle for instability is near 0.4 radians or 25°. At this point the PLM has an angular velocity of 0.12 radians/sec., giving the end of the PLM a velocity of 2.35 m/s or 5mph.

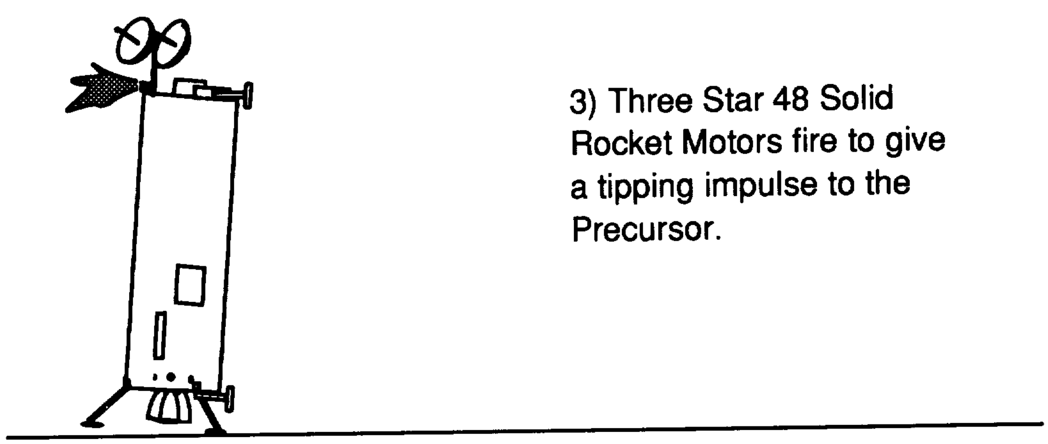
Two XLR-132 liquid rockets control the reorientation to horizontal and the soft landing criteria. Once the PLM has toppled and is pitching toward the lunar surface, the liquid motors begin to burn. The liquid motors perform a throttled burn for 9.3 seconds to keep the PLM at the same angular velocity. At the end of the 9.3 seconds, the PLM's cylindrical axis is 25° above the plane of the lunar surface. At this angle, the engines burn full bore for another 7.4 seconds, allowing the PLM to touch down on the lunar surface with no angular velocity. The deployment sequence is summarized in Figure 7-2 on the following two pages.



1) The Precursor starts the deployment sequence in vertical position



2) Four Habitat Levelling Legs extend in anticipation of horizontal deployment.



3) Three Star 48 Solid Rocket Motors fire to give a tipping impulse to the Precursor.

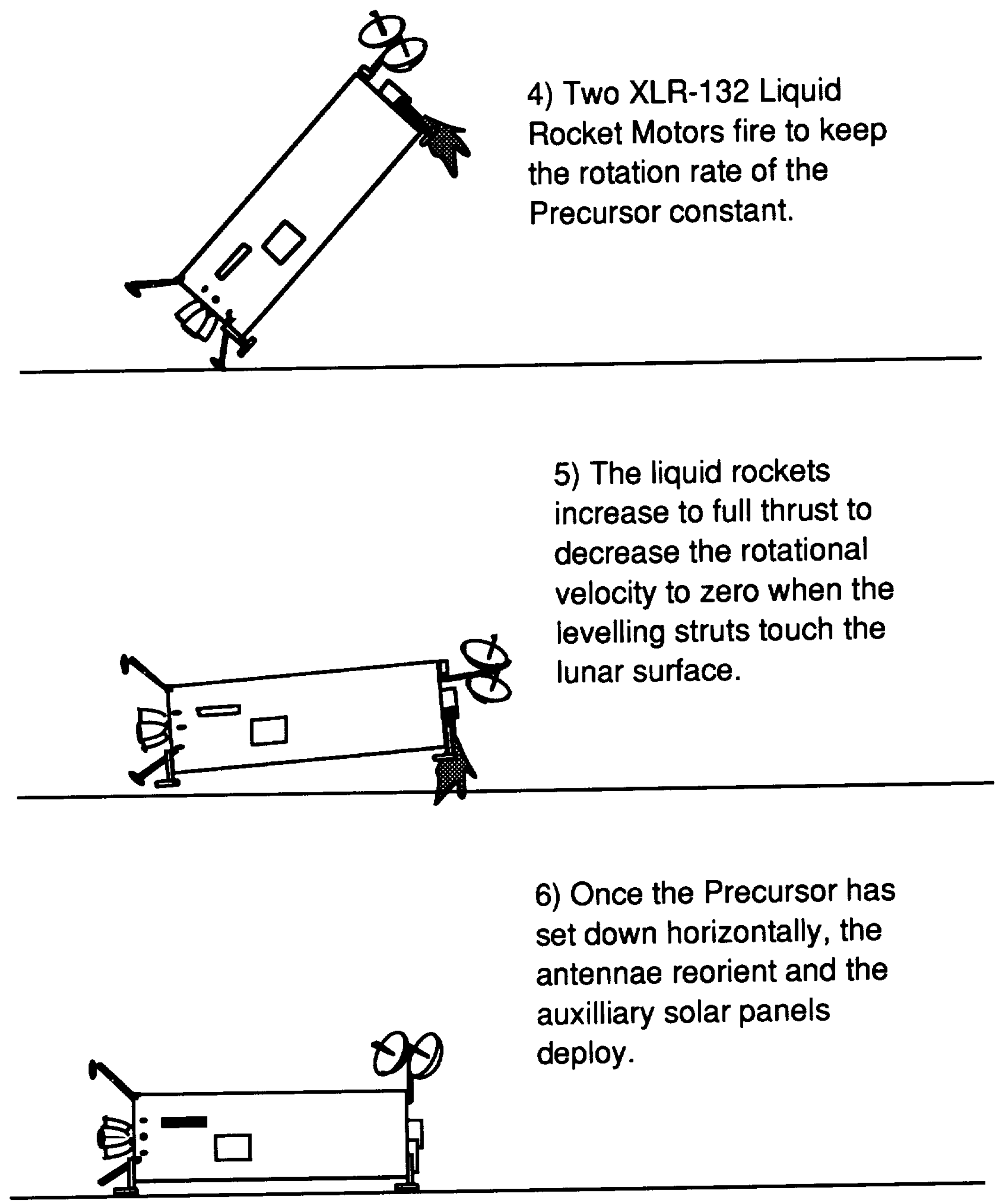


Figure 7-2
Precursor Deployment Sequence

7.1.3.4 Payload Deployment

Once the precursor lander has been reoriented, the PLM must deploy its auxiliary solar panels. These auxiliary panels are included to provide power for surface operations before personnel arrive. The constant power flux into the PV arrays runs the habitat and communications, allowing the SLURPP fuel energy to remain at full capacity. The two solar panels are folded in an accordion fashion into either side of the PLM. To open the panels, their deployment hatches open, and preloaded spring forces in the structure of the panels naturally allows them to unfold. Figure 7-3 displays the photovoltaic (PV) arrays deploying.

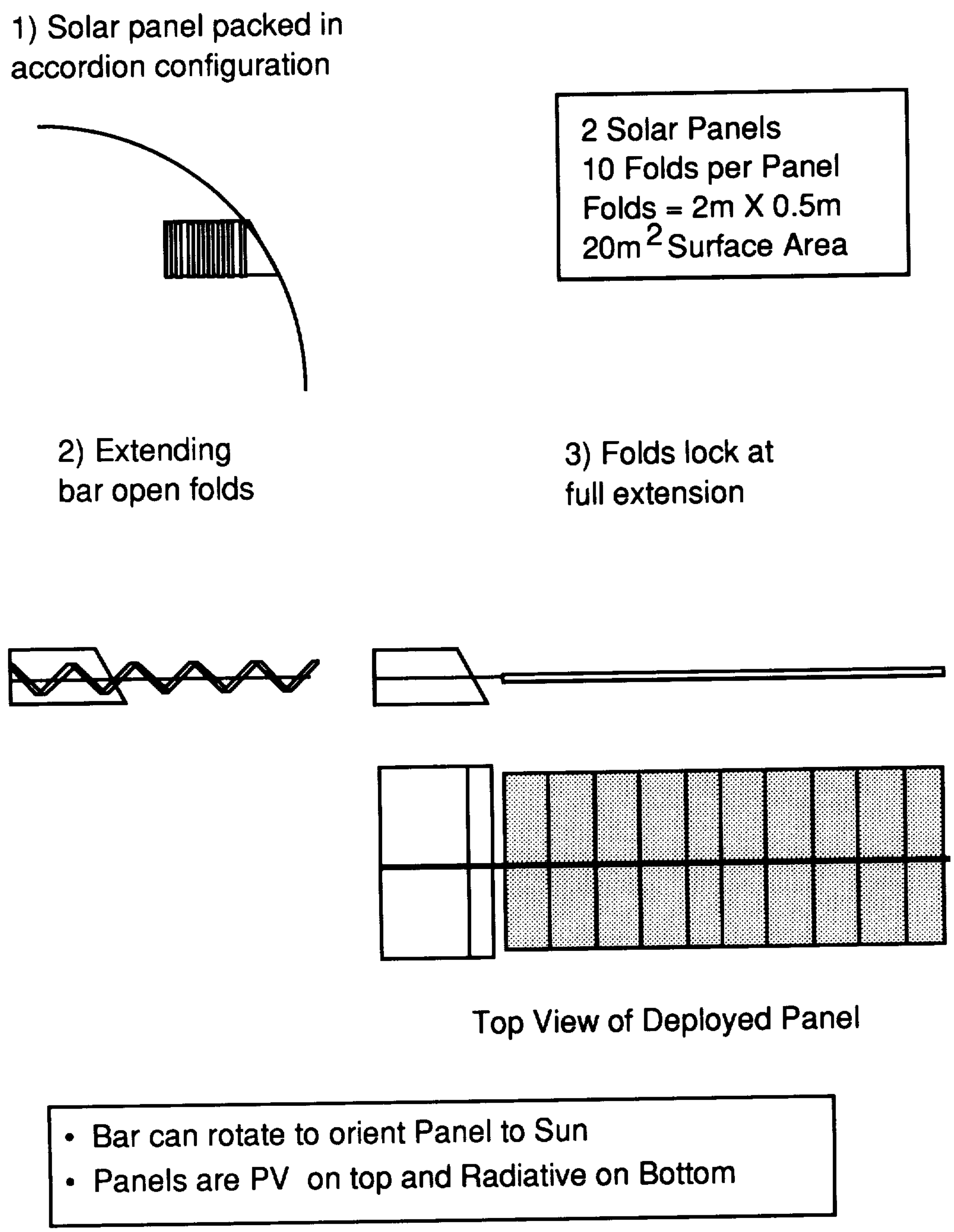


Figure 7-3
PV Array Deployment

7.1.3.5 Cargo Extraction from the PLM

The PLM cargo remains packaged in its Cargo Bay until personnel arrive on the Moon. The cargo is removed sequentially. First, the exit hatch is opened, and a rgang plank is positioned to facilitate egress. The lunar rover emerges first. It is pulled out partially, and its nose section is flipped out. Next, it is rolled completely out so that the rear section can be flipped out. These actions configure the rover for lunar duties.

The next cargo to be extracted from the PLM is the external SLURPP units. The PLM permanently houses the fuel cells, liquifiers, water bladder, Hydrogen dryer, and power conversion equipment. The SLURPP equipment to be removed and configured are the PV arrays, their structural and motor equipment, and cabling to the PLM.

The last major pieces of cargo in the PLM are the regolith bagger and conveyor. Both are packaged as parts and must be assembled. They are not configured in the PLM to be more volumetrically efficient. The astronauts assemble them after setting up the PV arrays.

7.1.4 Abort Options

The PLM supports abort options for the precursor mission. A successful abort results in 1) range/lunar landing zone safety, and 2) safing of spacecraft components, where possible. Safing during this phase of the mission implies the safe landing of the habitat module on the lunar surface. Ideally, landing of the habitat will occur in the primary landing zone; in the event that this is not possible, *any* safe landing of the habitat will potentially constitute a successful abort.

The PLM is double engine-out failure tolerant (1 out of 3 engines operable) for landing.

7.2 Stage Design

7.2.1 Configuration

7.2.1.1 External Configuration

The PLM is a cylindrical module measuring 6m in diameter and 9.25m high. The PLM/habitat system measures 20m high. The landing gear of the PLM extend the module another 2.24m further, but they tuck into the LBM and do not extend total height. See Section 4.1.1 of this Volume for a description of the space allotted in the LBM. The PLM attaches at its base to the LBM. The PLM attaches at its top to the lunar habitat. The PLM and lunar habitat are structurally integrated into one piece. A schematic of the undeployed PLM can be seen in Figure 7-4 showing its integration with other parts of the precursor vehicle.

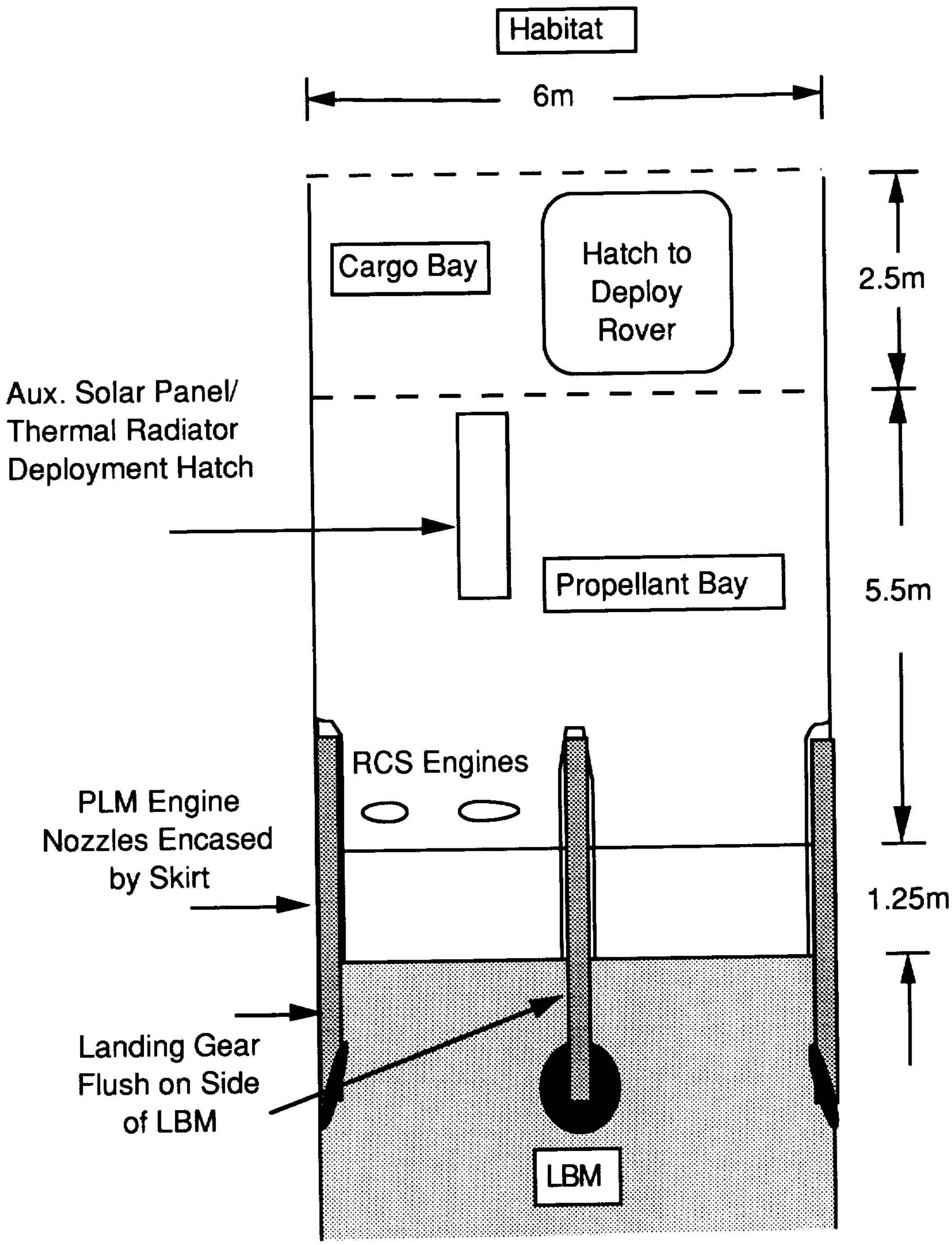


Figure 7-4
External, Undeployed Configuration of PLM

The PLM is divided into 4 sections. The lower 1.5m of the PLM is the RL10A-4 engine nozzles that are shrouded by a skirt that is discarding during staging from the LBM. This space also contains two of the habitat levelling struts and the four landing struts. The three landing struts spaced evenly around the cylinder. They measure 4.24m in length, extending from the end of the PLM rocket truss and overlapping the LBM. The next section is the Propellant Bay. The Propellant Bay houses the main engines except for their nozzles, the cryogenic fuel tanks, the lower RCS suite, guidance sensors, the Hydrogen dryer, reliquifaction units, and fuel cells. The Cargo Bay stores precursor payload including the Solar Lunar Power Plant (SLURPP) photovoltaic (PV) solar array units and cabling, the lunar excursion vehicle, the regolith bagger and conveyor, and miscellaneous cargo. The Cargo Bay has a side hatch measuring 2.25m by 2m to allow the payload to be removed and configured. After the Cargo Bay is the lunar habitat. The final section of the PLM at the other end of the habitat is the Nose Section. The Nose Section is affixed to the end of the habitat and resides under the nose cone during launch. The Nose Section includes the upper RCS suite, two of the habitat levelling struts, and the deployment engines. This basic layout of the lower portion of the PLM before staging was shown in Figure 7-4.

The PLM external configuration changes slightly for deployment. The deployed configuration of the PLM is seen from the side in Figure 7-5. First, the nose cone on the habitat is shed from the mission once it attains orbit. Second, the landing struts are opened and the RL10 nozzles are exposed when the PLM stages from the LBM and the nozzle skirt. The landing struts open to a 45° angle from the cylindrical axis, and are supported by bracing members. The deployed PLM lands vertically with its cylindrical axis parallel to the lunar surface. The PLM rests on the four landing struts; the RL10A-4 main engines remain 0.5m above the lunar surface. Footpads bearing spikes reside on the base of the struts for better stability on uncertain lunar terrain.

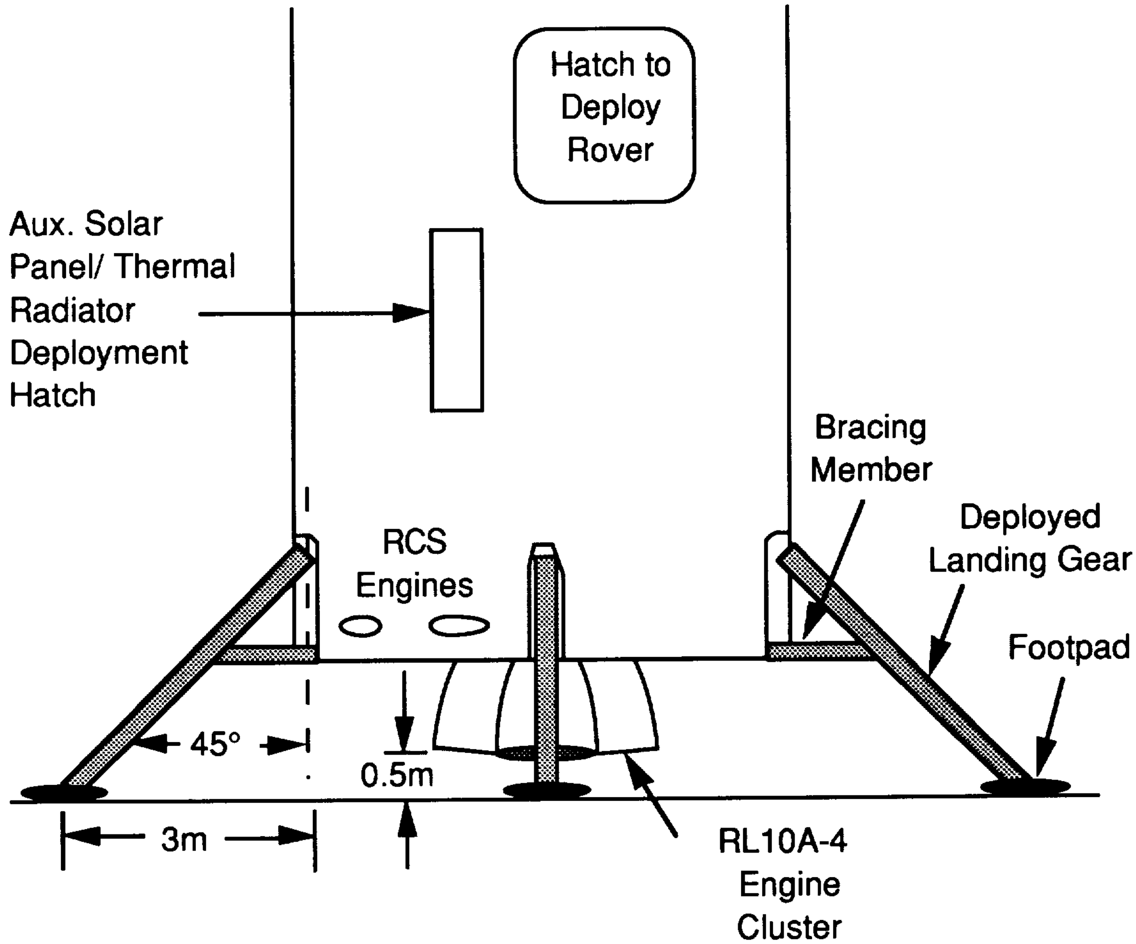


Figure 7-5
Side View of Deployed PLM

Figure 7-6 exposes additional details of the PLM by showing a bottom view of the deployed configuration. The equal spacing of the landing struts spaces them every 90° around the PLM. The bracing members add stability and strength to the struts for landing on uneven lunar soil. The RL10A-4 engines are also mounted 5° off the PLM's cylindrical axis. This off-centering choice is made to align the thrust closer to the center of mass. The line from the center of mass to the RL10A-4 engines is near 7° off the cylindrical axis. The 5° parameter allows the engine to gimbal from -2° to +6° from the center of mass instead of the symmetric ±4° for an ideally oriented engine. If one or two engines fail, the remaining ones can gimbal -2° to +6°. When all three engines are working, they are gimballed 4° to be only 1° off of the centerline. This 1° drops the thrust of the PLM by cos(1°) to 99.98% of ideal. If an engine is shutdown, the thrust of the PLM drops by cos(5°) to 99.6% of ideal. The bottom of the PLM also has two of the four habitat support legs. The other two are in the Nose Section on the other end of the habitat.

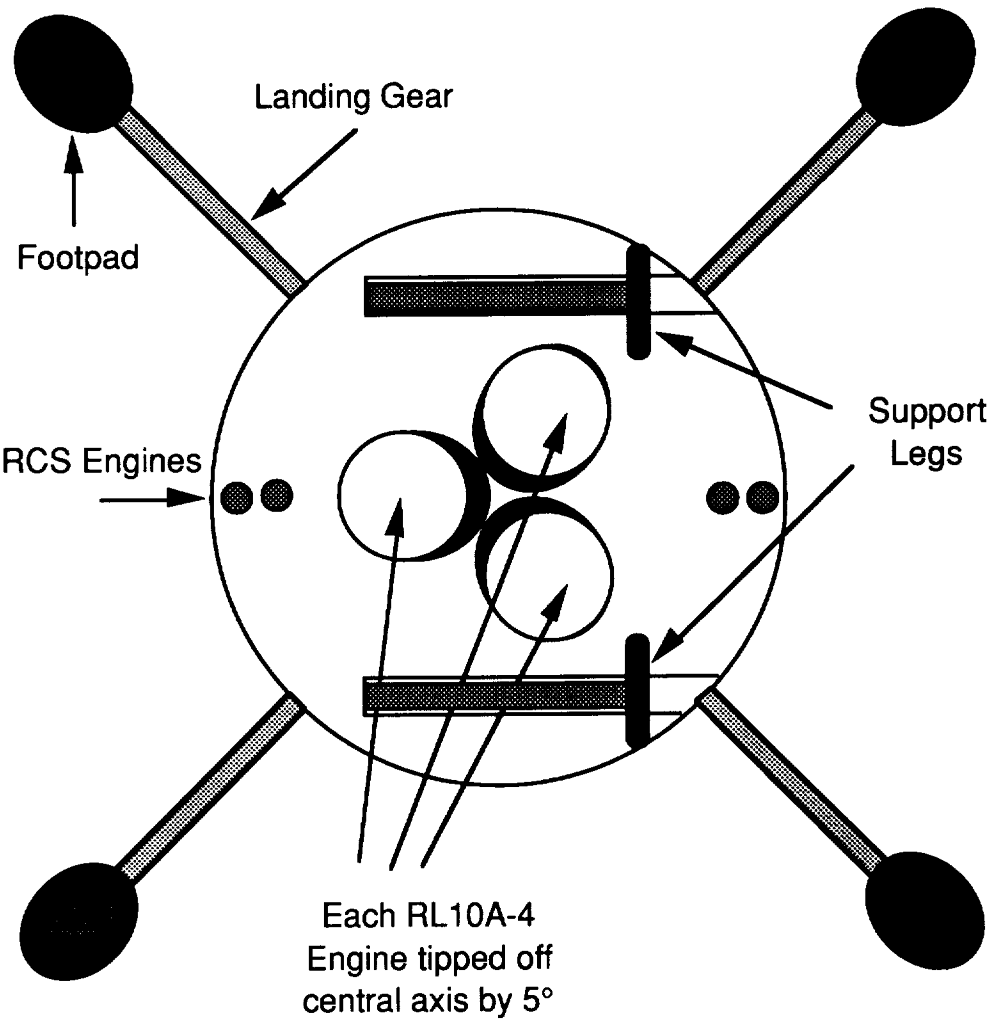


Figure 7-6
Bottom View of PLM with Deployed Struts

7.2.1.2 Internal Configuration

The general internal configuration of the PLM is summarized in Figure 7-7.

The Propellant Bay contains many subsystems. The rocket support truss, the RCS engines, RCS tanks and piping, the engine portion of the RL10A-4s, the Helium pressurization tanks, and guidance sensors are are all in the base of the Propellant Bay seen in Figure 7-9 as the Lower RCS suite cross-section. The rocket support truss distrubutes the loads from engine thrusts, landing gear, and the launch vehicle evenly to the structure. The truss also provides a surface to gimbal the RL10A-4 engines from. The cryogenic fuels of Hydrogen and Oxygen are each stored in a pair of spherical tanks, making a cluster of four tanks. The diameter of each Hydrogen tanks is 3.00m, including 20cm of Kapton cryogenic insulation around the entire sphere. The diameter of the Oxygen tanks is 2.4m including the same 20cm ofinsulation as the Hydrogen. Ringing the propellant tanks are SLURPP subsystems including reliquifiers, a Hydrogen dryer, fuel cells, and auxiliary solar panels/radiators. There is a wall separating the Propellant and Cargo Bays. The Cargo Bay contains equipment for surface operations including the lunar rover, SLURPP PV array equipment, and the regolith bagger and conveyor. The water bladder that stores the water made by the fuel cells rests in the bottom of the Cargo Bay beneath the rover floor. The water is saved to be converted back into elemental form using energy from the PV arrays. The Hydrogen tanks and Oxygen tanks feed both the SLURPP fuel cells and the RL10A-4 main engines.

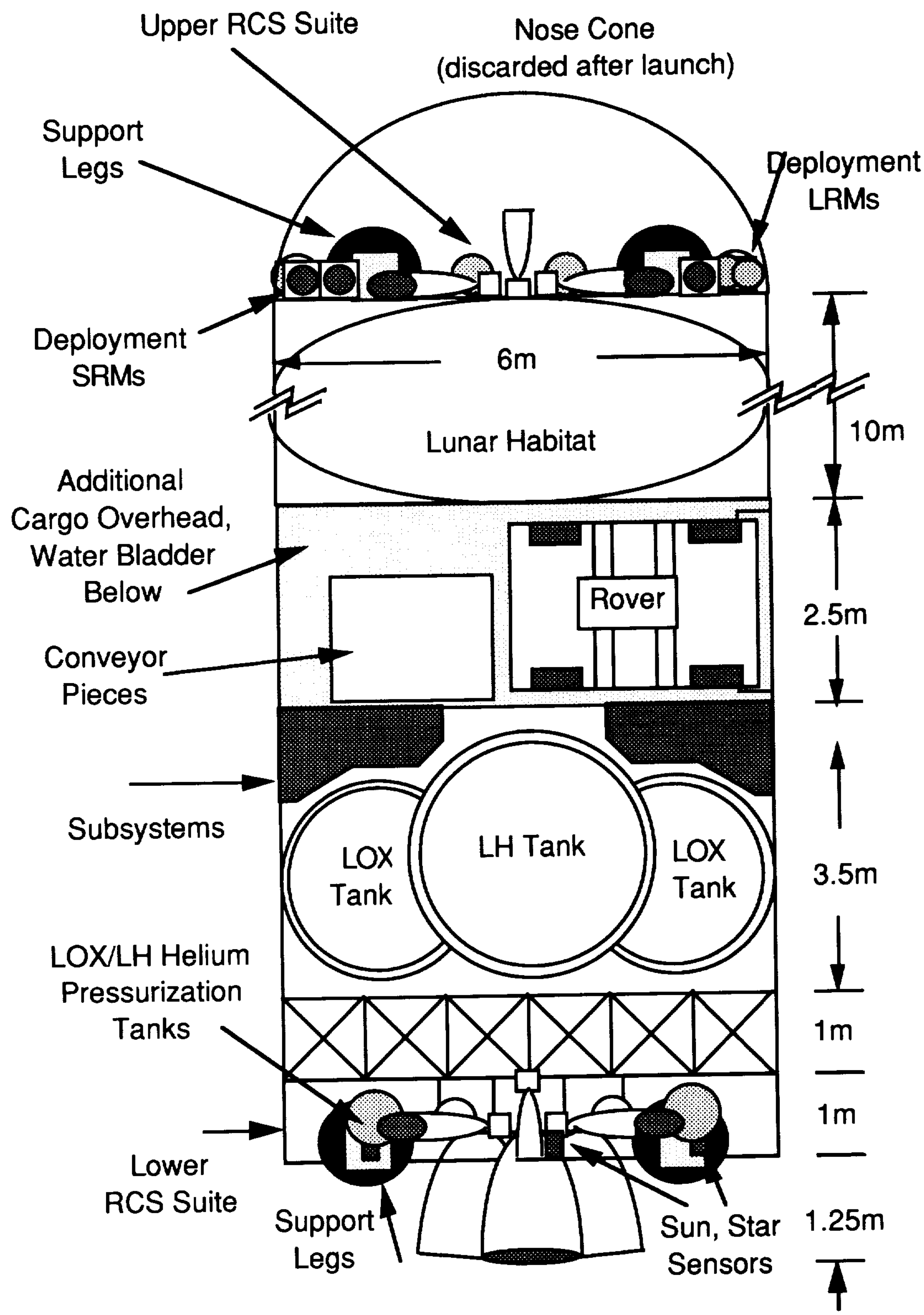


Figure 7-7
Internal Top View of PLM

Figure 7-8 exposes another angle of the PLM, and clarifies the spatial relation of the subsystems further.

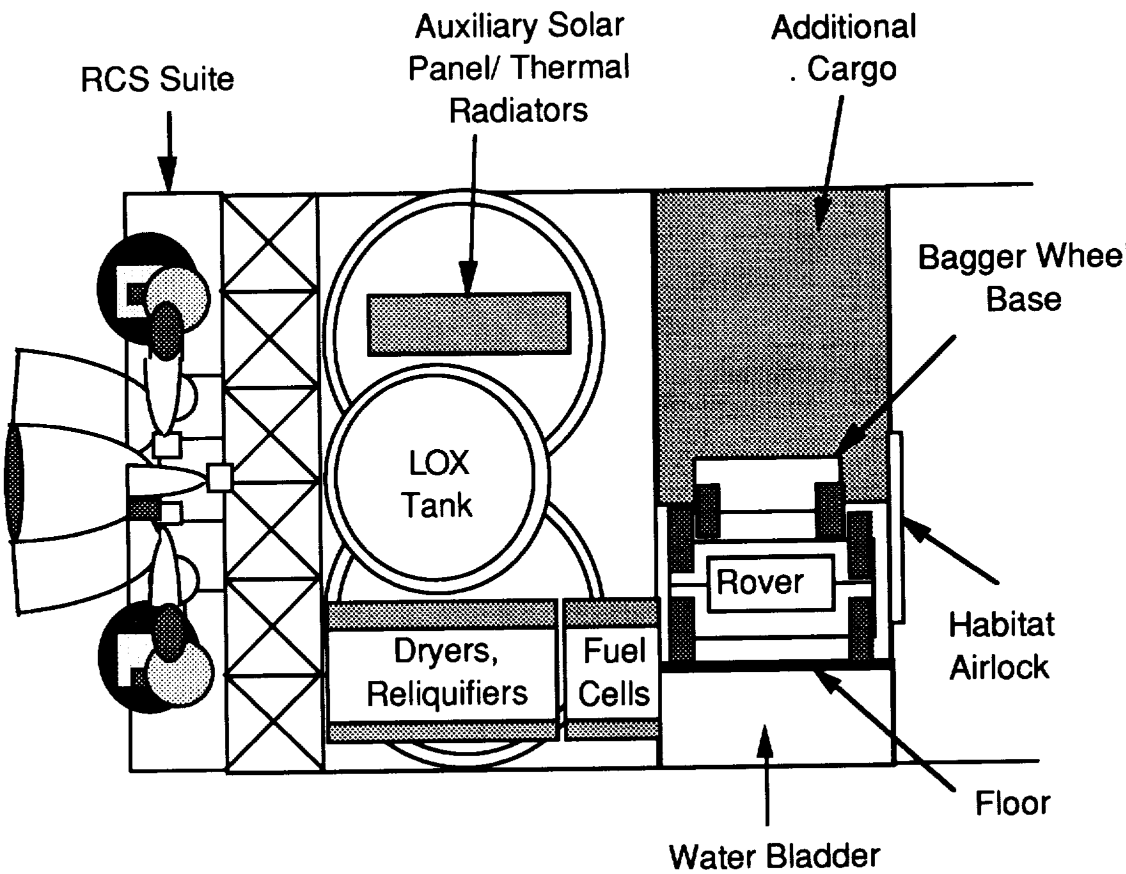


Figure 7-8
Internal Side View of PLM

The lower Reaction Control System (RCS) suite resides at the base of the Propellant Bay. It consists of eight thrusters: two redundant pairs aligned along the cylindrical axis and the other four in the parallel plane. The RCS system is positioned around the main engines. The entire lower RCS suite is in the general shape of a pineapple ring. The RCS thrusters are spaced in triples to provide movement in the x-direction, y-direction, and rotation around the z-axis (cylindrical axis). The six thrusters are fed by tanks of Monomethyl Hydrazine and Nitrogen Tetroxide. Each tank has its own Helium pressurization tank. These eight thrusters work in conjunction with the eight in the upper RCS suite at the end of the habitat for spacecraft control during the entirety of flight. The upper RCS suite has an identical thruster arrangement. The RCS suite also houses some additional components: Helium tanks for pressurizing the main fuel tanks, sun sensors, star sensors, an altitude radar for guidance, and two struts for levelling the habitat once it is deployed. Figure 7-9 clarifies the volume apportionment in the RCS ring that encircles the RL-10 engines.

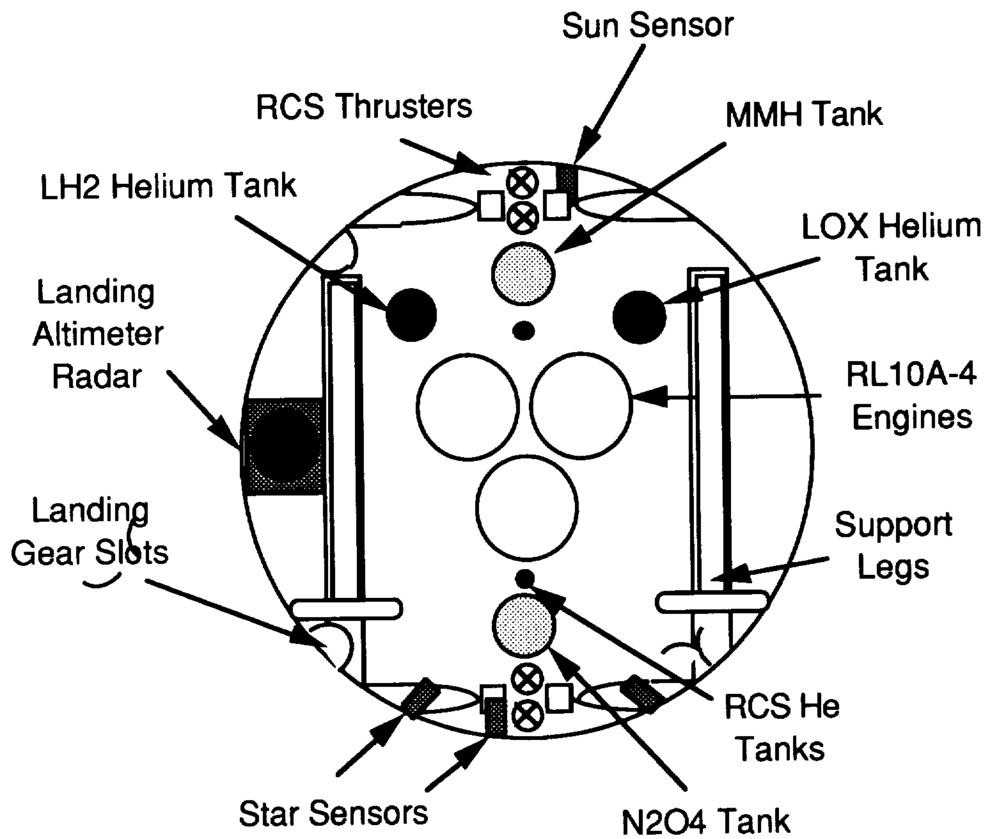


Figure 7-9
Cross-Section of RCS Suite

Above the RCS suite in the Propellant Bay, subsystems are placed around the propellant tank cluster. The SLURPP fuel cells are housed to either side of the lower Hydrogen tank toward the Cargo Bay end. These cells provide power to each mission module during flight as well as providing full power to the surface mission. The reliquifaction units and the Hydrogen dryer are in line with the fuel cells, but further down in the PLM. Further around the perimeter, above the Oxygen tanks are a symmetric pair of Auxiliary Solar Panels. These solar panels gather energy to perform environment control for the habitat and to run the reliquifaction units that keep the SLURPP fuels from boiling away. These solar panels provide enough power so that the fuel energy of SLURPP is not squandered. Once personnel arrive, the main solar panels will be deployed and energy will be available to support additional lunar missions. The back side of the solar panels are lined with radiative materials for thermal control. A cross-section of the Propellant Bay of the PLM is shown in Figure 7-10.

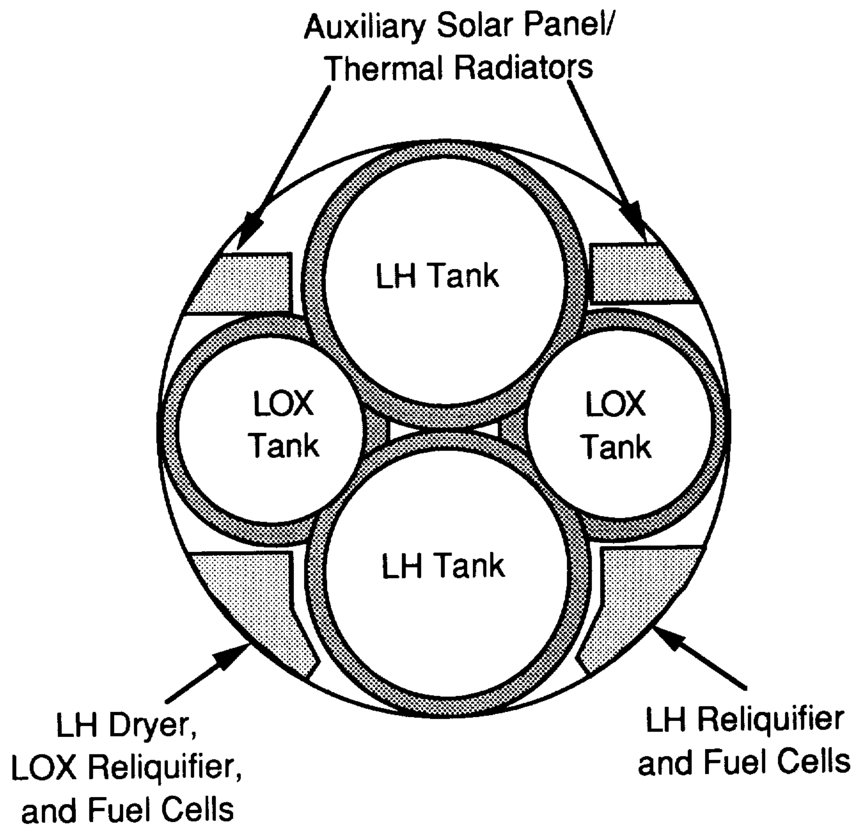


Figure 7-10
Cross-Section of PLM Propellant Bay

The Cargo Bay stores a variety of equipment: a lunar rover, a regolith bagger, a regolith conveyor, regolith support structure, SLURPP PV arrays, and array support structure. The parts of this cargo that do not fit in the Cargo Bay are stored inside the habitat. The lunar rover sits on a floor in the Cargo Bay so that it can readily roll out through the side hatch to the lunar surface. A gangway is provided for the rover to roll from the hatch down to the surface and for the astronauts to remove the rest of the cargo. The floor covers over a water bladder that rests on the bottom of the Cargo Bay. When the fuel cells provide power, they combine Oxygen and Hydrogen to for water. The water is stored in the bladder until solar energy can be utilized to perform hydrolysis and split the water back into its elementary components. Behind the rover are the main components of the coonveyor belt. Above the rover is the wheel base of the regolith bagger. Above and around these main cargo pieces are addditional cargo, mainly the regolith support structure. This volumetric arrangement could be partially seen in Figures 7-7 and 7-8. Figure 7-11 illustrates the cross-section of the Cargo Bay.

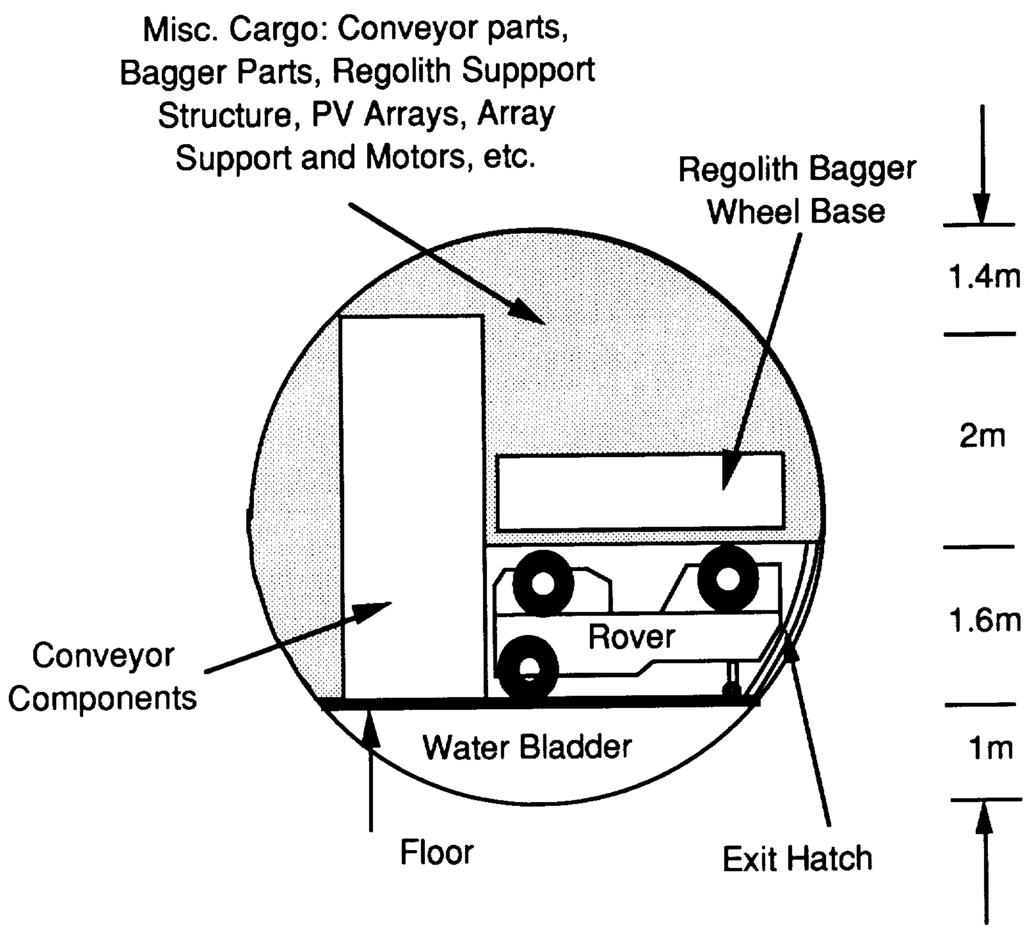


Figure 7-11
Cross-Section of PLM Cargo Bay

The last active part of the PLM is the Nose Section containing the upper RCS suite, two habitat support struts, the communications antennae, and the deployment engines. Both the solid and liquid deployment rockets are affixed to the habitat by truss structures. The two struts, along with the two identical struts on the other end of the PLM allow the habitat to remain above treacherous terrain such as rocks and keep it orientated properly. The upper RCS suite is identical to the lower RCS suite in thruster configuration. Figure 7-12 provides an end view of the Deployment Module and the spatial relation of the major components on that end of the habitat. A more detailed description of the deployment procedure can be found in Section 7.1.3.3.

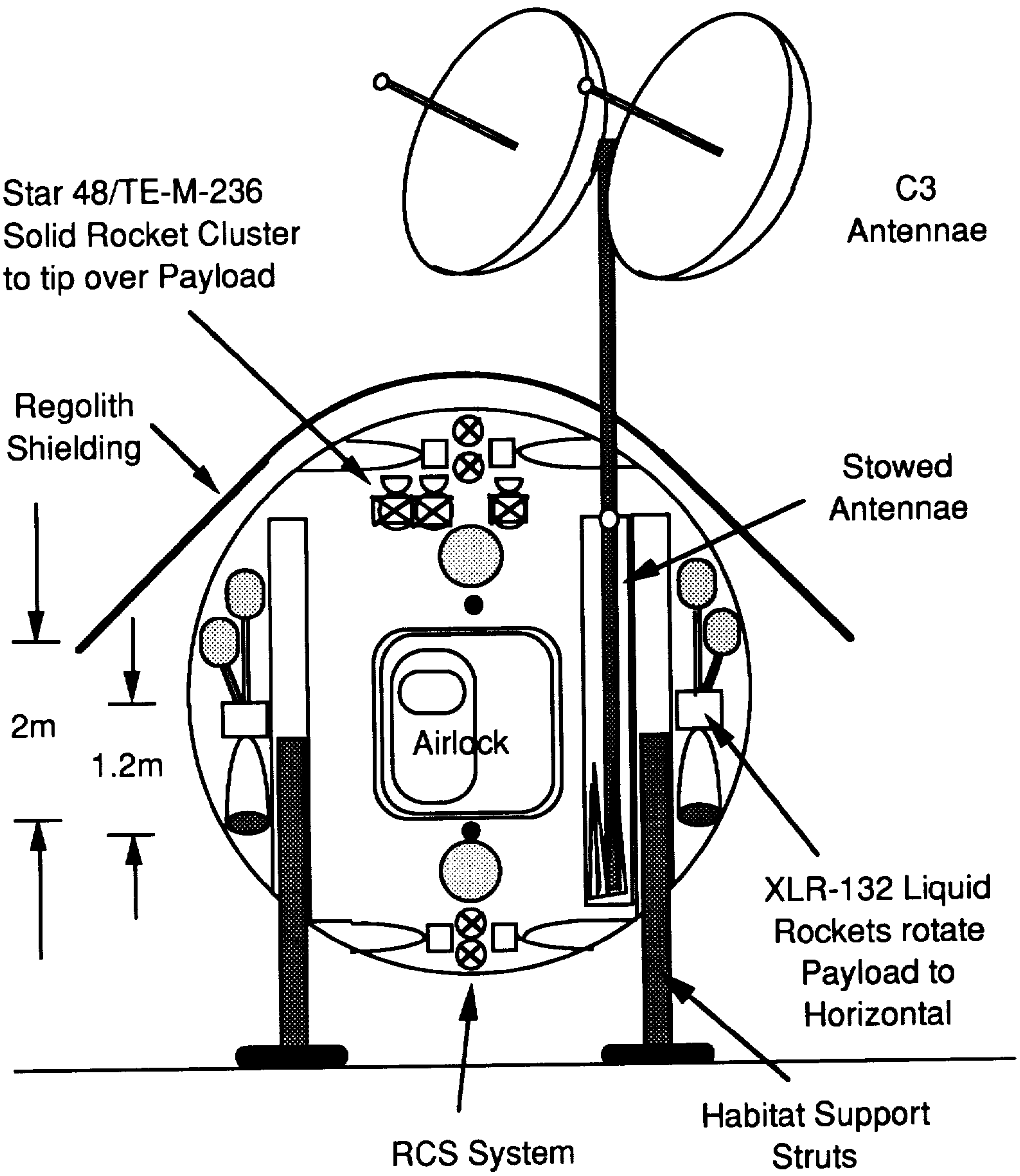


Figure 7-12
End View of Deployment Package

Figure 7-13 is a side view of the Deployment Package. The figure shows the relation of the Nose Section to the habitat and how it fits inside the launch nose cone.

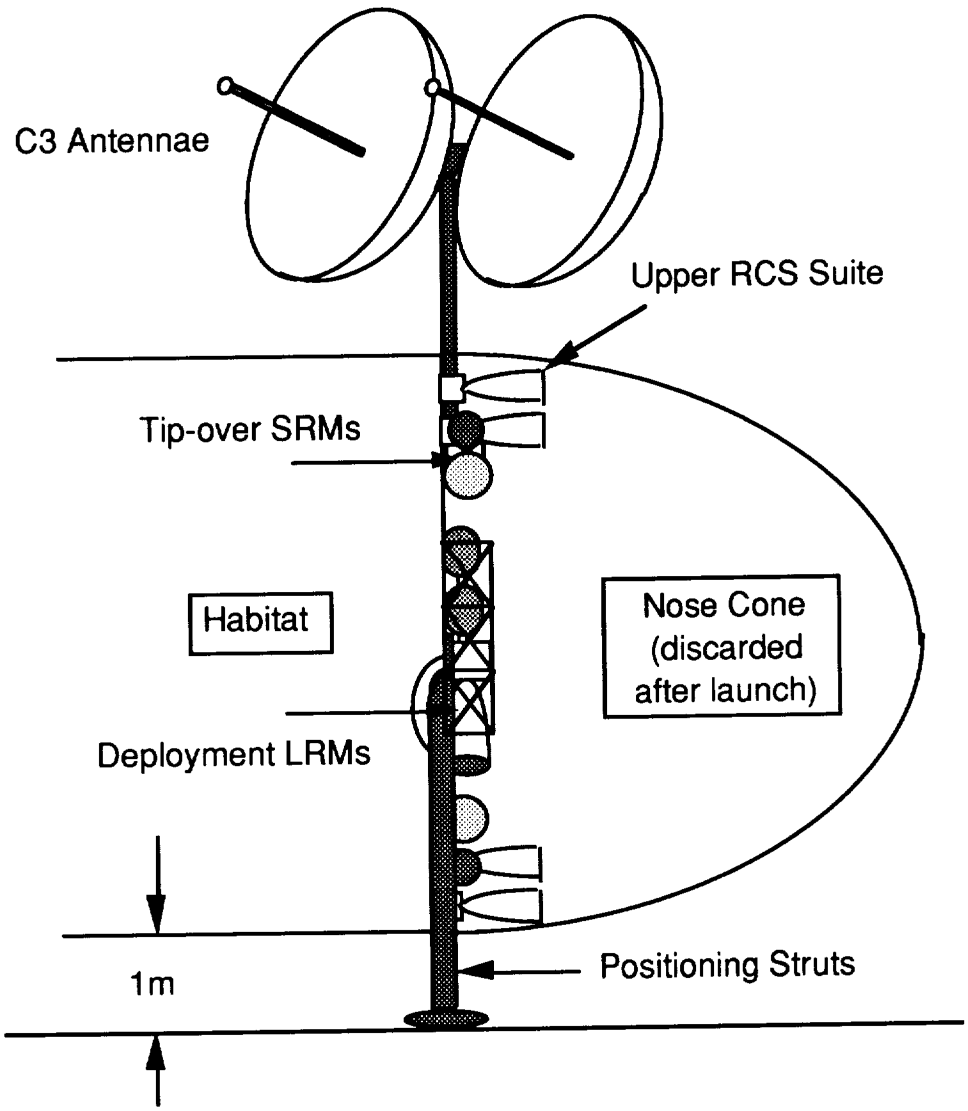


Figure 7-13
Side View of Deployment Package

Once the PLM has reoriented to a horizontal position, a few additional components are deployed. The Auxiliary Solar Panels/Thermal Radiators extend through hatches in the side of the PLM. The solar panels are parallel to the lunar surface, but can swivel to orient to the solar radiation. The communications antennae reorient to face the Earth. The three landing struts hang off the side of the PLM after they transfer support duties to the levelling struts. When personnel arrive, the exit hatch is opened and the exit ramp is positioned to facilitate removal of the PLM cargo. Cargo removal is discussed in Section 7.1.3.5. These changes in external configuration of the PLM are shown in Figure 7-14.

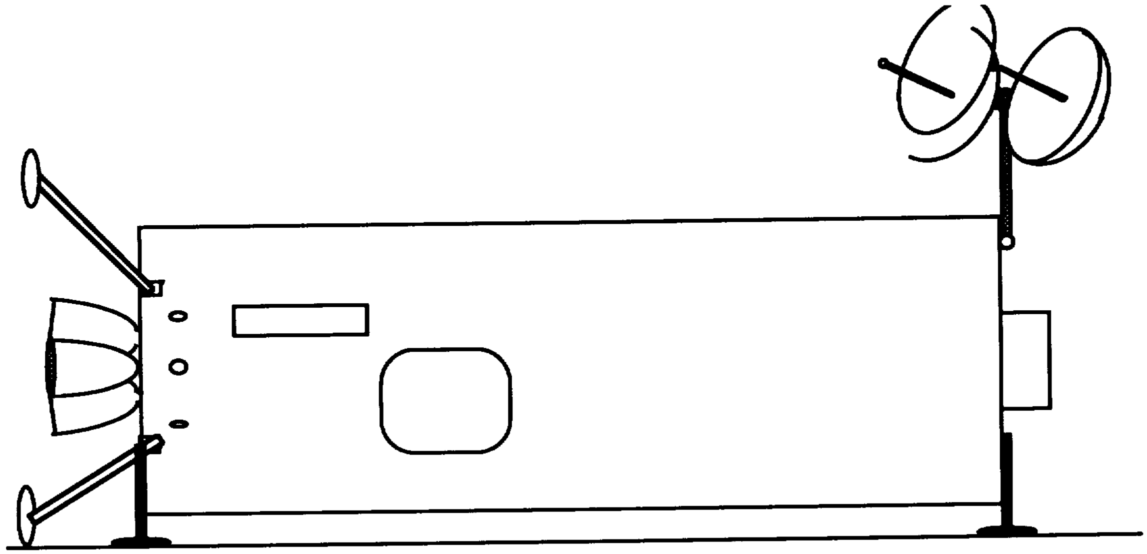


Figure 7-14
External Deployed PLM Configuration

7.2.2 Vehicle Interfaces

This section documents the interfaces for the PLM.

7.2.2.1 Lunar Braking Module

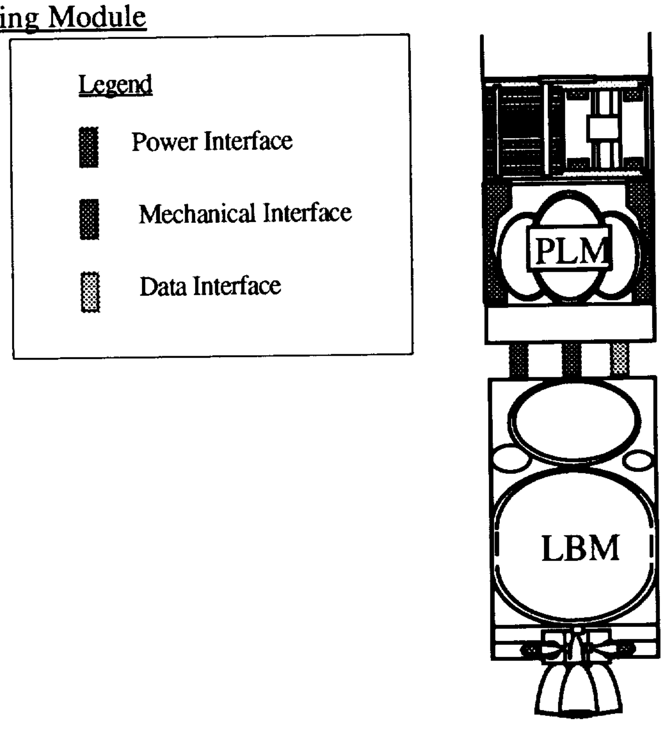


Figure 7-15
PLM/LBM Interface

7.2.2.1.1 Mechanical Interface

The mechanical interface between the LBM and PLM consists of explosive bolts for stage separation. The landing struts on the PLM nestle into grooves down the side of the LBM. To make the struts flush with the LBM skin, they are recessed and have covers that keep the LBM skin continuous.

7.2.2.1.2 Data Interface

The data interface is a database between the LBM and PLM which transmits LBM status to the computers in the Habitat. The data link also provides a connection between the LBM and PLM for command and engine control.

7.2.2.1.3 Power Interface

The power interface between the LBM and PLM connects the fuel cells in the PLM to all subsystems in LBM.

7.2.2.2 Habitat

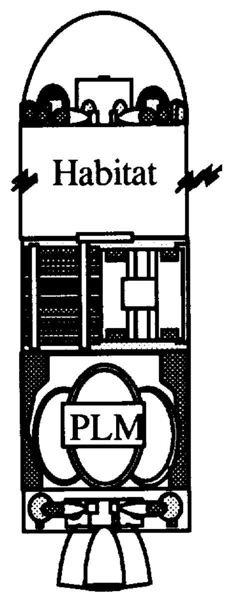


Figure 7-16
PLM/Habitat Interface

The interfaces for the PLM and the habitat are not shown in the figure to emphasize that the PLM and the habitat are contained in the same shell.

7.2.2.2.1 Mechanical Interface

The mechanical interface between the PLM and the habitat consists of an airlock since the PLM will not be pressurized but also needs to be accessible by the crew.

7.2.2.2.2 Data Interface

The data interface consists of a fiber optic database that connects the PLM to the main computers in the habitat. The database will monitor the power, control, and status of the PLM and habitat.

7.2.2.2.3 Power Interface

The power interface between the PLM and the habitat will supply power from the fuel cells in the PLM to all subsystems.

7.2.2.3 Nose Cone

The interface between the PLM and the nose cone is described in Figure 7-15. The interfaces for the PLM and the nose cone are not shown in the figure to emphasize that the PLM and the nose cone are contained in the same shell.

7.2.2.3.1 Mechanical Interface

The mechanical interface consists of explosive bolts to separate the nose cone from the habitat before the stack is tipped over.

7.2.2.3.2 Data Interface

The data interface will transmit the command of the deployment engines.

7.3 Subsystem Design

7.3.1 PLM Configuration

The Payload Landing Module, or PLM, will take the lunar habitat and all necessary set-up equipment down to the lunar surface after the LBM is ejected (Figure 7-17). The main body of the PLM itself is a semi-monocoque cylinder. This Primary Hull is designed to take the brunt of the axial and lateral launch accelerations, as well as the bending stresses after the structure is deployed horizontally.

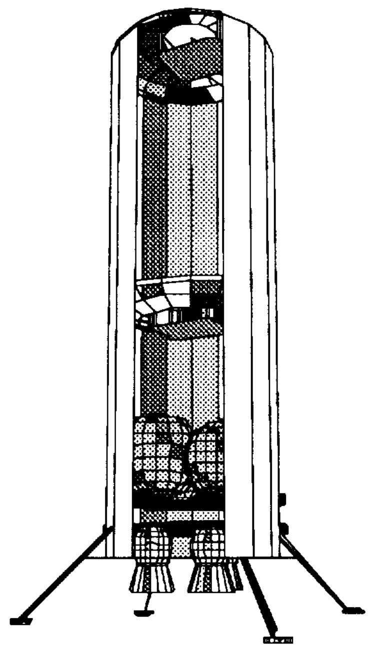


Figure 7-17
PLM Stage—Cutaway View (landing)

The PLM has two sets of support structures to buffer it from the lunar surface. The landing gear is composed of 4 landing legs which deploy just before LBM separation. These legs provide stability and cushion the impact at touchdown. After the PLM has landed, it will tilt to a horizontal position (Figure 7-18). In this deployed state, the vehicle will rest on four support legs, which will serve as a permanent supports for the lifetime of the lunar base.

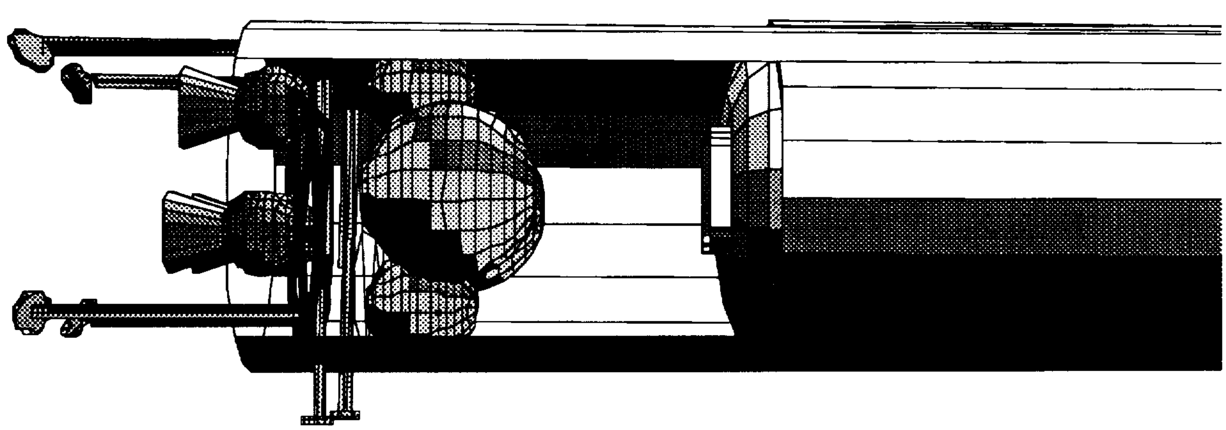


Figure 7-18
PLM Stage—Cutaway View (deployed)

The remainder of the PLM stage structures are internal and are grouped into three sections for discussion (Figure 7-19). At the base of the PLMis the Propulsion Section where the rocket motors and propellant tanks are attached to the Rocket Truss. Above this section is the cargo bay used for storing the solar arrays, lunar rover, regolith support structure, and various other supplies and machines necessary for set-up of the lunar base. At the top of the PLM rests the BioCan lunar habitat, where the crew will live for the 28-day mission.

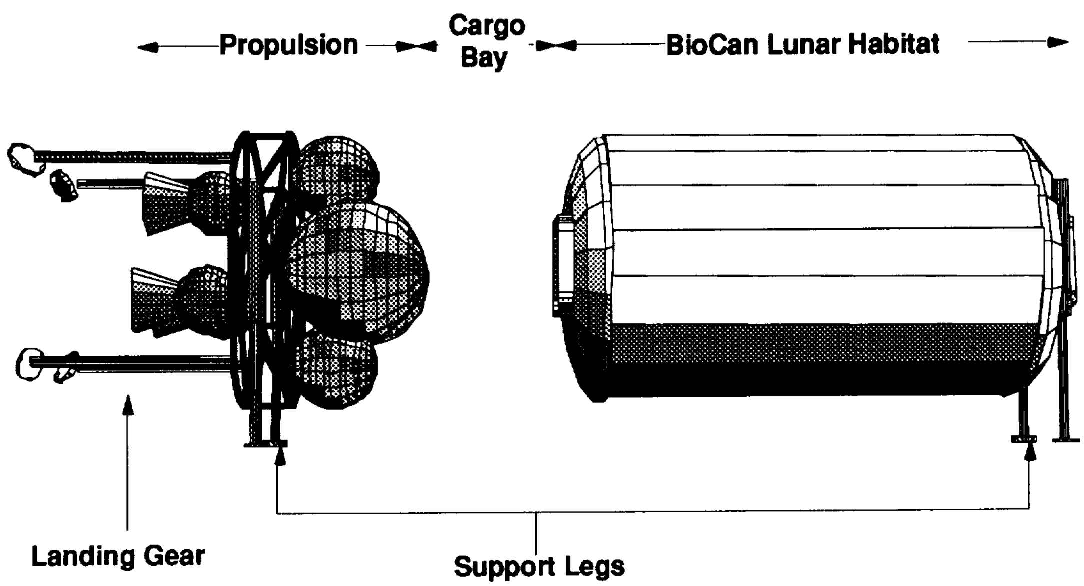


Figure 7-19
PLM Stage—Internal Structures

7.3.1.1 PLM Primary Hull

Load Criteria

The PLM stage is expected to withstand launch accelerations of up to 3.5g axially and 2.5 laterally. In its horizontal position after deployment, the stage must endure the bending loads due to its own weight as well as the regolith shielding which will cover it.

Configuration

The main body of the PLM itself is a semi-monocoque cylinder or radius 6m and length 19m. This Primary Hull is designed to take the brunt of the axial and lateral launch accelerations, as well as the long-term bending stresses once the structure is deployed horizontally. There are 12 stringers and 18 frames in the design (Figure 7-20). The frames make up the largest portion of the total framwork mass, due to the large stresses induced by lateral accelerations and regolith shielding. Specifications are given in Table 7-4.

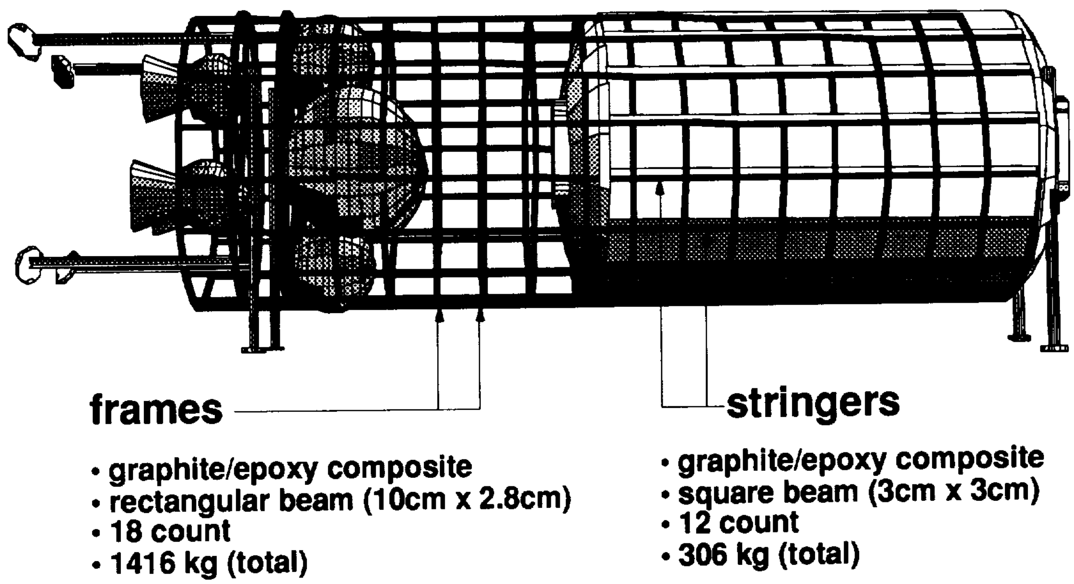
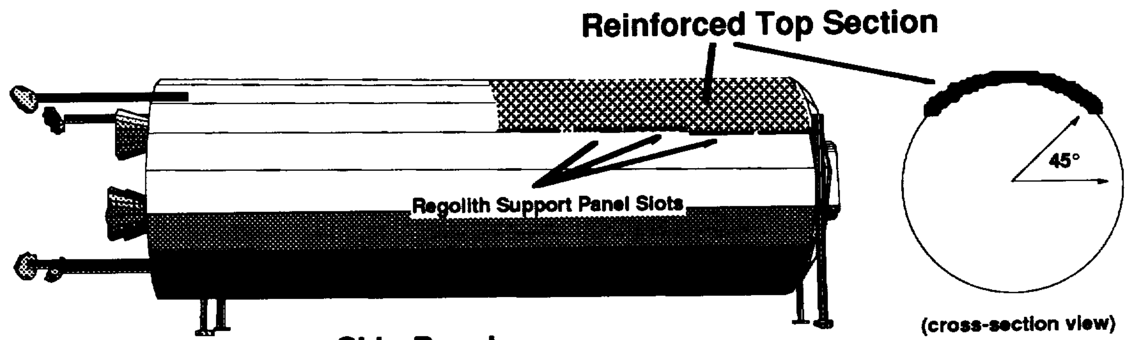


Figure 7-20
PLM stage—Framework

The framework is covered with thin, curved skin panels. These bolt-on panels are removable to allow access to the BioCan pressure vessel and other internal structures for inspection and repair. The major portion of the skin is composed of 0.5 mm composite panels. The primary forces on these panels are aerodynamic. The portion of the skin which will support regolith above the BioCan is composed of reinforced aluminum panels, 2mm thick (Figure 7-21). Along the bottom edge of this reinforced section are several horizontal slots. These slots are important in the assembly of the Regolith Support Structure, to be discussed later in this chapter. Table 7-5 at the end of this section contains a summary of the Primary Hull statistics.



Skin Panels

- 1.12m axial width
- 1.57m radial width
- 3.0m radius of curvature

	Material	Thickness	Count	Weight
Normal	graphite/epoxy composite	0.5 mm	177	232 kg
Reinforced	aluminum	2.0 mm	27	263 kg
Total			204	495 kg

Figure 7-21
PLM stage—Skin Panels

Summary Specifications

Table 7-5: PLM Primary Hull Specifications

PLM Body:			
Body Diameter	6.0 m	**Stringers: (graphite/epoxy composite HTS |0|)**	
Body Radius	3.0 m	Stringer Cross Section Type	square
Propellant Section Height	5.69 m	Stringer outer radius	0.030 m
Cargo Bay Height	2.50 m	Stringer inner radius	0.030 m
BioCan Height	11.00 m	Number of Stringers	12
Total Body Length	19.03 m		
Panels:		**Frames: (graphite/epoxy composite HTS |0|)**	
Panel axial width	1.119 m	Frame Cross Section Type	rectangular
Panel radial width	1.571 m	Frame height	0.10 m
Panel radius of curvature	3.0 m	Frame width	0.028 m
		Frame Spacing	1.12 m
Normal Panels (graphite/epoxy composite |0| HTS)		Number of Frames	18
Number of Normal Panels	177		
Normal Panel Thickness	0.0005 m		
Reinforced Panels (aluminum 2024-T36)			
Number of Reinforced Panels	27		
Reinforced Panels Thickness	0.002 m		
Total Number of Panels	204		
MASS ESTIMATES			
Mass of Stringers	294 kg		
Mass of Frames	1,429 kg		
Mass of Panels	495 kg		
Lander Body Mass Subtotal	1,968 kg		
Joints & fittings allowance	25%		
Primary Hull Mass(empty)	2,460 kg		

7.3.1.2 Ground Support

7.3.1.2.1 Landing Legs

Load Criteria

The landing gear for both the Precursor and Piloted landing vehicles is identical. The legs are required to support the entire weight of the vehicle (about 26 metric tons) under a landing shock of 0.6 g. The horizontal velocity component is expected to be negligible at touchdown. The craft is expected to be reasonably stable, yet for the case of the Precursor mission, it is desired to topple the PLM by a set of solid rocket motors at deployment time. Therefore, two of the landing legs are expected to support the entire weight of the vehicle for a brief period during deployment. In addition, the soft, uncertain lunar regolith necessitates some sort of landing feet to prevent excessive sinking of the legs into the surface.

Configuration

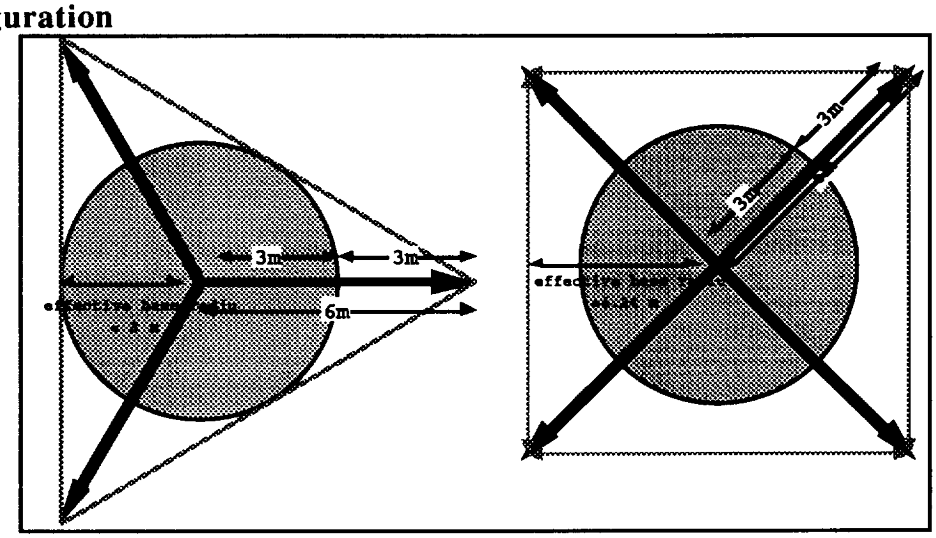


Figure 7-22

Effective Base Radius Comparison for 3-leg and 4-leg Cases

The quality of stability of a landing gear configuration with a circular spread can be expressed by its effective base radius, or the length of the moment arm generated by the landing legs in the direction most susceptible to toppling (Figure 7-22). The effective base radius is determined by the number, length, and angle of the landing legs. The tripod and four-leg configuration were considered most seriously for this project. The four-leg configuration was chosen over the tripod because of its favorable mass to effective base radius ratio.

Each landing leg makes an angle of 45° to the surface. The angle was chosen as a trade-off between the larger angles with large bending moments and smaller angles with less stability.

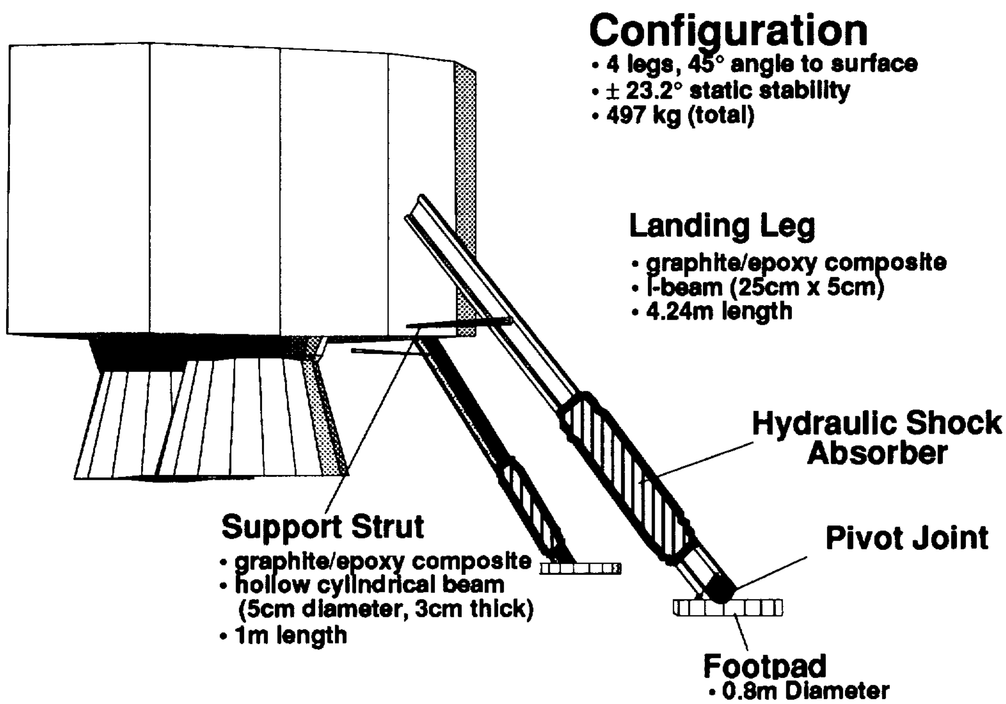


Figure 7-23
Landing Gear Configuration

Each landing leg consists of a main beam, support strut, and footpad (Figure 7-23). The main beam is a composite I-beam, equipped with a hydraulic shock absorber to cushion impact at touchdown. The I-beam configuration was chosen to more efficiently react the large bending moments in the vertical direction. The hydraulic shock absorber was chosen over a crushable balsa shock absorber used in the Apollo moon missions due to its reusability. If the initial landing site proves unsatisfactory for some reason, it may be possible to use the remaining fuel on board to relocate. The footpad is attached to the main beam via a pivot joint, which allows the footpad to accept any surface angle upon landing. This pivoting is also necessary to accomodate the toppling motion of the PLM during deployment. The joint is spring-centered to prevent awkward footpad angles upon initial contact with the surface. The support strut acts to reduce the moment arm of the main beam at its connection with the Rocket Truss. Its construction is a hollow cylindrical composite beam. A screw-action motor pushes the support strut outwards to deploy the landing leg (Figure 7-24). The support strut is much smaller and lighter than the main beam since it is not expected to see large moments, but only axial loads. The sizes and masses of the various components are given in Table 7-6 at the end of this section.

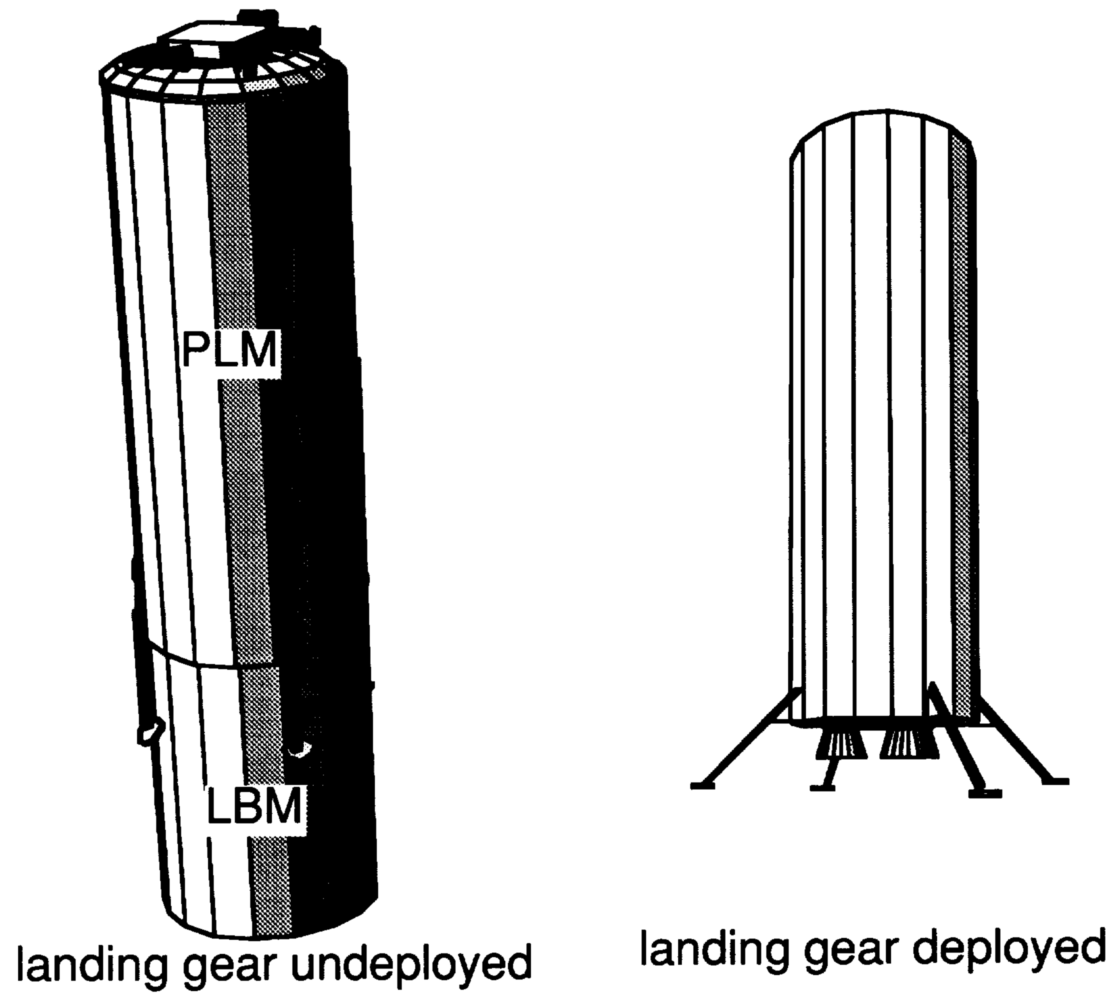


Figure 7-24
Landing Gear Deployment

Summary Specifications

Table 7-6: Landing Gear Geometry & Mass Estimate

GEOMETRY		MASS ESTIMATE	
Leg Length	**4.24 m**	Footpad mass	22.5 kg
Leg Angle	45°	Support bar mass	0.7 kg
Ground Clearance	2.00 m	Mass of Main beam	45.8 kg
Support Bar length	1.00 m	Joints & Fittings	35%
Stage radius	3.00 m	Motors & Misc	100 kg
Effective base radius	4.24 m		
Footpad thickness	0.03 m	**Total Landing Gear Mass**	**497 kg**
Footpad radius	0.40 m		
Number of Legs	4	**STABILITY**	
		Center of Mass	9.9m above surface
		Stable Angle (deg)	**23.2°**

7.3.1.2.2 Support Legs

Load Criteria

The Support legs keep the entire PLM structure from touching the lunar surface in order to prevent thermal conduction and also to level the structure and provide a comfortable living environment for the crew. During the deployment procedure, these legs must carry the entire weight of the PLM through the landing shock experienced after toppling. For the lifetime of the habitat, these legs must carry not only the weight of the entire stage, but also the weight of the lunar regolith shielding which will cover the habitat . These items will be discussed in more detail in the next section, *Regolith Support Structure.*

Configuration

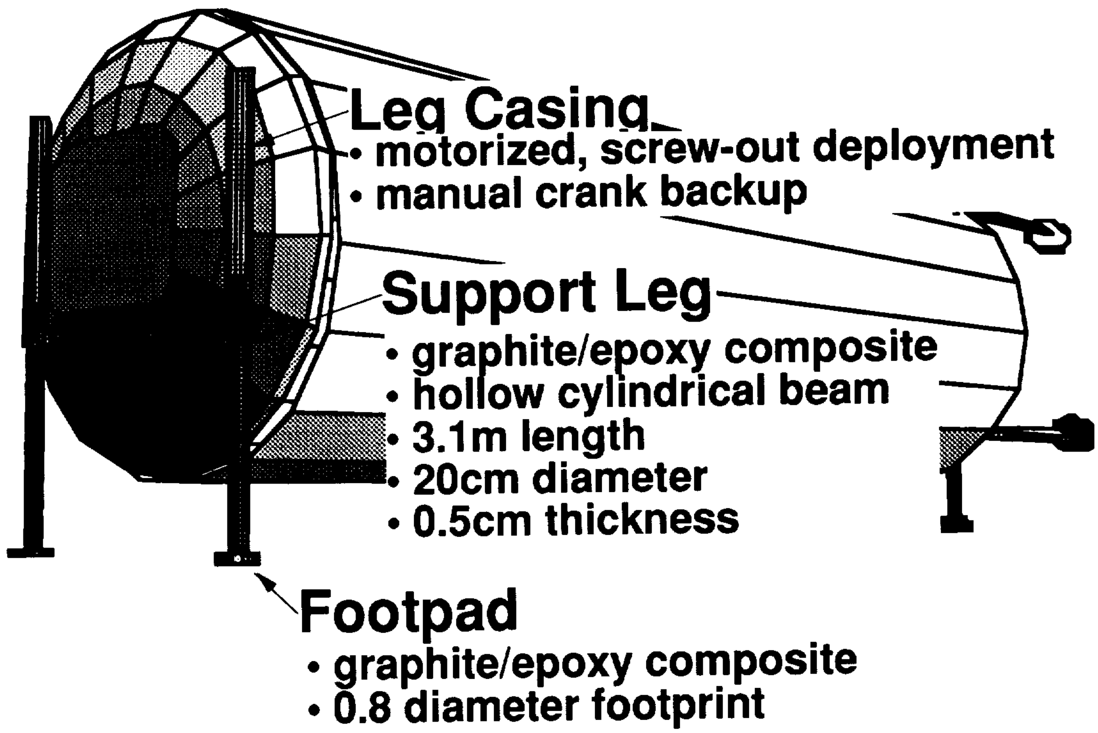


Figure 7-25
Support Leg Configuration

There are four support legs on the PLM stage (Figure 7-25). Each is a hollow, cylindrical beam made of graphite/epoxy composite. These four legs extend out of their casings by mechanical screw-action motors to full length shortly before toppling deployment. Each leg can also be deployed by manual cranking as a redundant backup in case of motor failure. At maximum extension, the ground clearance on a hard surface is one meter. At the end of each leg is composite footpad with a 0.8 diameter footprint. After making sure that the hull will not be breached by underlying rocks, the PLM will be slightly lowered to make crew access and regolith shield construction easier. Table 7-7 presents a summary of the support leg specifications.

Summary Specifications

Table 7-7: Support Leg Geometry & Mass Estimate

GEOMETRY		number of legs	4	**MASS ESTIMATE**	
body radius	3.00m	Leg outer radius	0.100m	Single leg mass	14.23kg
distance from center	2.87m	Leg inner radius	0.095m	Foot mass	14.98kg
distance from bottom	2.12m	Foot Radius	0.4m	Leg mass subtotal (4 legs)	117 kg
in-case allowance	0.50m	Foot Thickness	0.02m	Casing/Extension Motor Allowance	150%
ground clearance	1.00m				
Leg length	3.12m			Total Support Leg Mass	292 kg

7.3.1.3 Propulsion Section

Load Criteria

The propulsion section must transfer the thrust from the three RL-10 rocket engines to the rest of the vehicle and store the liquid hydrogen and liquid oxygen propellants to be used in the engines. In addition, the propellant tanks will be used to store the fuel for the fuel cells which will power the lunar base while the solar cells are ineffective during the 14-day lunar night. The propellant tanks will be under 340,000 Pa of internal presssure in addition to the dynamic pressure of the contents during launch acceleration.

Configuration

The propellant tanks are mounted on the top of the Rocket Truss. The two hydrogen tanks and two oxygen tanks are mounted side by side. The configuration of 4 spherical tanks side-by-side was chosen to reduce the height of the vehicle. The hydrogen tanks decide the height of this propellant section because of their greater size (Figure 7-26 & Table 7-8). Each tank is a graphite/epoxy composite pressure vessel with a wall thickness is 0.5 mm. The tanks are covered externally with insulation for the cryogenic contents. This thickness is 16.3 cm for the hydrogen tanks and 10 cm for the oxygen tanks. A half millimeter of steel lining on the interior of the tanks prevents the cryogenic contents from reacting adversely with the composite tank walls.

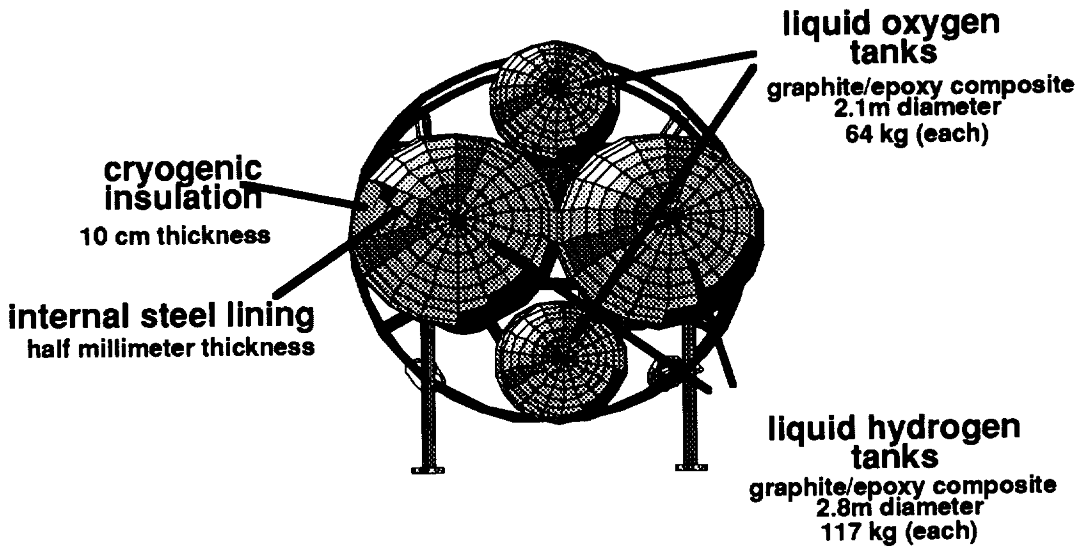


Figure 7-26
PLM Propellant Tanks

Summary Specifications

Table 7-8: PLM Propellant Section Specifications

Configuration		**Oxygen Tanks**		**Hydrogen Tanks**	
Truss Mass	250 kg	Oxygen Mass	8780	Hydrogen Mass	1270
Tank Mass	362 kg	Oxygen Volume	7.4951	Hydrogen Volume	18.7817
Tank Truss, Piping, Fittings	350 kg	Oxygen Tank Geometry	spherical	Hydrogen Tank Geometry	spherical
Insulation Mass	458 kg	Number of Oxygen Tanks	2	Number of Hydrogen Tanks	2.0000
Engine Mass	501 kg	Oxygen Tank Radius	0.9636	Hydrogen Tank Radius	1.3088
Section Dry Mass	**1921 kg**				
		Oxygen Tank Wall Thickness	0.0005 m	Hydrogen Tank Wall Thickness	0.0005 m
Total Section Height	**5.70**	Oxygen Tank Insulation Thickness	0.10 m	Hydrogen Tank Insulation Thickness	0.163 m
		Oxygen Steel Lining Thickness	0.0005 m	Hydrogen Steel Lining Thickness	0.0005 m
		Oxygen Tank Mass w/fittings	64 kg	Hydrogen Tank Mass w/fittings	117 kg
		Oxygen Tank Insulation Mass	57 kg	Hydrogen Tank Insulation Mass	172 kg
		Oxygen Tank Mass w/fittings & insulation	121 kg	Hydrogen Tank Mass w/fittings & insulation	289 kg

7.3.1.4 Cargo Bay

The cargo bay is a vacancy located between the BioCan pressure vessel and the propellant tanks of the PLM stage. The structural components of this section consist mainly of fitting and shelves to store the solar panels, regolith support structure, lunar rover, and construction machinery during the flight. No new calculations are performed specifically for this section, but two features need to be mentioned briefly—the access hatch and the gangplank.

7.3.1.4.1 Hatch

An access hatch exists on the side of the PLM stage to facilitate unloading of the cargo bay. This section is not pressurized, so the hatch need not be airtight. However, once deployed in the horizontal position, stress concentrationd can arise in the primary hull near the hatch when in is opened. This necessitates a "beefing up" of the frame surrounding the hatch to compensate.

The hatch for the cargo bay is shaped identically to the wall section that is replaces. The hatch opens by sliding up and away on two side rails, much like a typical garage door.

7.3.1.4.2 Gangway

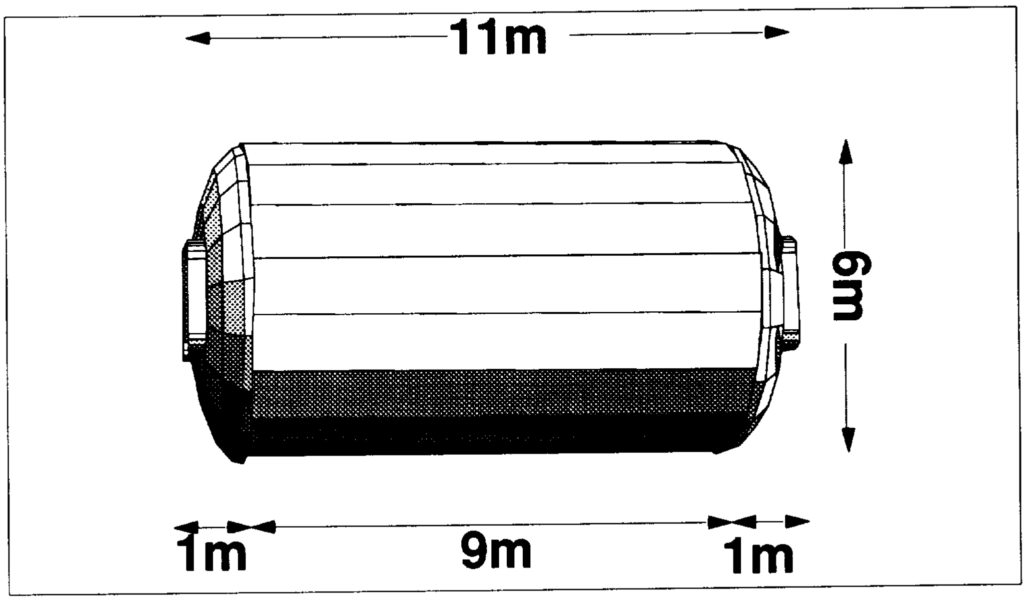
The crew will need a convenient way to get large, heavy objects in and out of the cargo bay. A gangplank has been chosen for this purpose. The lunar rover will drive down the gangplank, and the solar arrays and regolith support structure will also be carried across its length. In the interests of modularity, and because the expected loads are about the same order of magnitude, this gangplank is identical to one of the regolith support structure panels, discussed in Section 2.2.5.6, *Regolith Support Structure*. The gangplank must be located at an easily accessible location from the outside of the PLM, since the internal airlock of the BioCan may not be openable until the cargo bay is sufficiently unloaded to allow the airlock door to swing outward into the cargo bay. The gangplank slides out from the side of the PLM hull just under the cargo bay hatch.

7.3.1.5 BioCan Lunar Habitat

Load Criteria

The structure for the BioCan lunar habitat is expected to endure a 35000 Pa internal atmospheric pressure. It is also expected to endure the axial loads and lateral accelerations of launch on its walls and internal structures. It is not expected to experience the bending stresses present after deployment to the horizontal position, since most of these forces are taken by the PLM Primary Hull. A certain degree of thermal protection is needed in order to protect the habitat from the extremes of the lunar environment.

Configuration



BioCan Lunar Habitat

- 35,000 Pa cylindrical pressure vessel
- Aluminum
- 3:1 elliptical endcaps
- dual airlocks
- vacuum cavity insulated

Figure 7-27
BioCan Configuration

A configuration having at least two exit hatches is necessary in case of fire or other emergency. The cylindrical payload area of the launch vehicle puts constraints on the shape and size of the structure. A cylindrical configuration was chosen for the habitat section of the PLM stage (Figure 7-27). The cylinder body has a radius of 2.9 m and attaches to the inside of the frames of the Primary Hull. The BioCan itself is primarily an aluminum pressure vessel with wall thickness of 2mm. The elliptical endcaps have a 3:1 ratio, and extend another meter past the nine meter cylindrical body on each side. From the end of each endcap, the total length of the BioCan is 11 meters. A rectangular airlock exists on each side, situated in the endcaps. Table 7-9 at the end of this section shows the geometry and mass estimate for the BioCan pressure vessel.

Summary Specifications

Table 7-9: BioCan Geometry and Mass Estimate

GEOMETRY		**MASS ESTIMATE**	
Cylinder Diameter	5.8 m	Material	Aluminum
Cylinder Radius	2.9 m	Mass of Internal Structures	6669 kg
Cylinder Length	9 m	Basic Structure Mass	1245 kg
End Cap Ellipse Ratio	3:1	Airlock & Hatch Allowance	800 kg
End Cap semi-minor axis	1 m	Joints & Fittings	35%
		Total BioCan Structural Mass	**2760 kg**
Total BioCan length	11 m	Total BioCan Mass (full)	9429 kg
Skin Thickness	0.002 m		

7.3.1.6 PLM Stage Specifications Summary

Table 7-10 summarizes the information provided in this section on the PLM structure.

Table 7-10: PLM Specifications Summary

GEOMETRY	
Ground Clearance	2.00 m
Propulsion Section Height	5.53 m
Cargo Bay Height	2.50 m
Biocan Height	11.00 m
Total PLM Height w/out legs	**19.03 m**
Total PLM Height w/legs	**21.03 m**

MASS ESTIMATES	
Primary Hull	2549 kg
Landing Gear Mass	497 kg
Support Leg Mass	292 kg
Propulsion Section Mass (dry weight)	1921 kg
Biocan Mass (unfurnished)	2729 kg
Total PLM Structural Mass	**7987 kg**

7.3.2 Propulsion

7.3.2.1 Primary Propulsion System

The primary propulsion system of LBM stage is shown in Figure 7-28. It consists of three RL10A-4 engines rated at 92,518 N nominal thrust and operating each at a 5.5:1 mixture ratio of oxidizer to fuel. The net positive suction head (NPSH) required by the engine turbopumps is provided by pressurizing the vehicle propellant tanks with helium gas at 272 atm. Propellants are delivered to the main engine turbopumps through feed ducts from the vehicle propellant tanks. The feed ducts contain flex joints to accomodate engine gimbaling and are overwrapped with a three-layer, double aluminized Kapton radiation shield.

The primary propulsion engines run on a bipropellant combination of liquid oxigen oxidizer and liquid hydrogen fuel. There are four spherical propellant tanks; two of them store the oxidizer and two of them store the fuel. The tanks are constructed of a thin steel core overwrapped with pre-stressed graphite composite fibers and a 20 cm layer of aluminized Kapton insulation. The diameters of the oxidizer and fuel tanks are a 1.76 m and 2.78 m, respectively.

Pneumatically actuated prevalves located at the propellant tank outlets provide series redundant backup for the engine inlet shuttof valves. A parallel set of pyro valves and solenoid valves upstream of the pneumatic actuation control solenoid valves provides two-failure tolerance against inadvertent opening of the engine inlet shutoff valves. The pyro valves will be fired open after the LBM stage is deployed a safe distance from the PTLI stage. The system also has manual fill and drain valves to load propellant and pressurant gas into the system, as well as additional manual valves for system leak checking on both sides of the pyro-isolation valves and regulators. Check valves insure that the fuel and oxidizer can never mix anywhere in the system, except in the engine. Finally, pressure transducers, filters, temperature sensors, and line and component heaters are provided to ensure proper subsystem operation. A mass distribution of the entire propulsion system is given in table Table 7-11.

The fuel tanks carry significantly more propellant than is necessary for landing. Well over half of the propellant is for the SLURPP fuel cells that provide power during the lunar night. Because these fuel cells will be stationed on the Moon for a long period before the piloted mission arrives, there is a thick layer of passive insulation around the tanks to minimize cryogenic fuel boil-off. The cryogenic propellant boil-off is also avoided by using the SLURPP reliquifaction units.

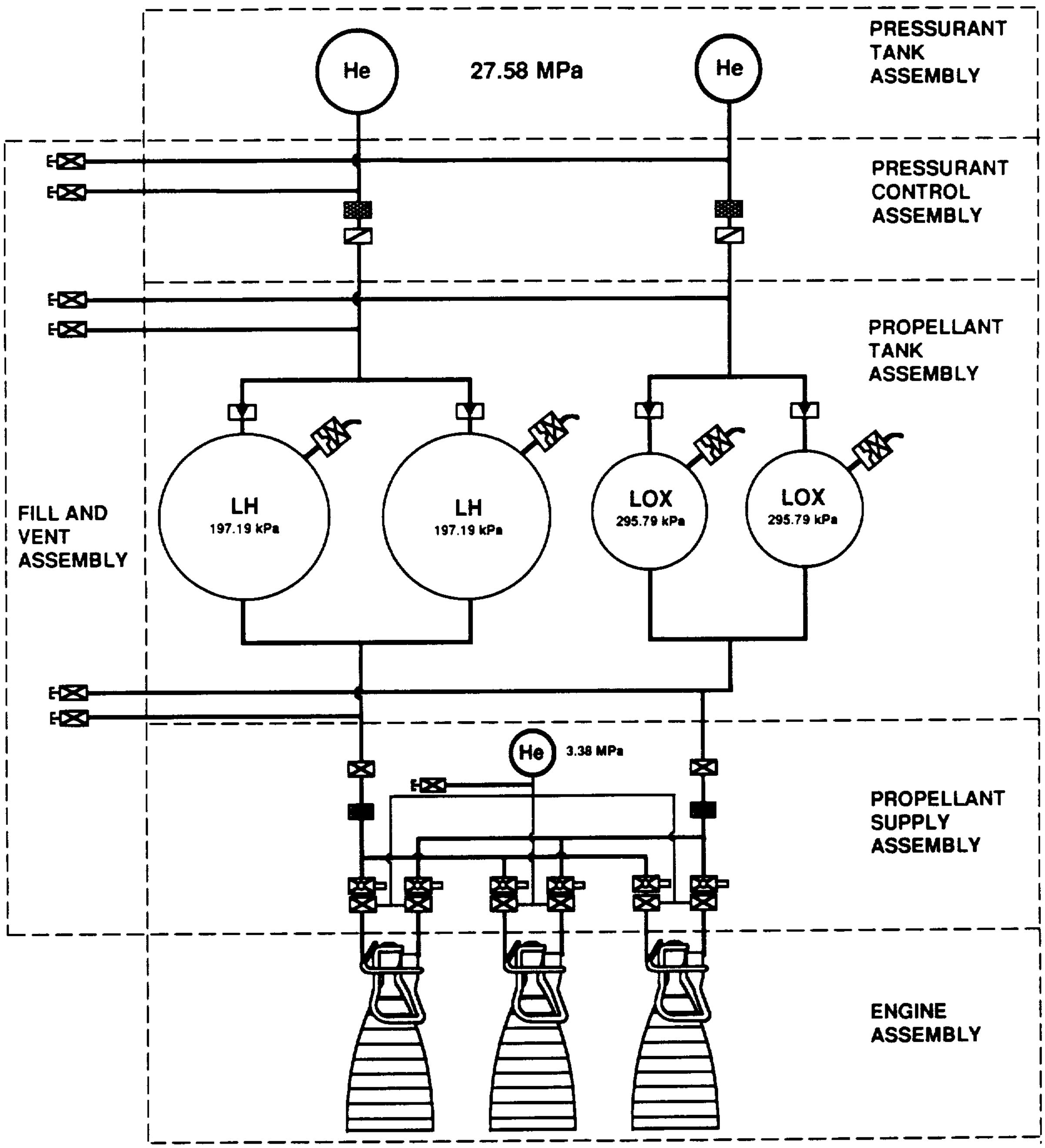


Figure 7-28
PLM Primary Propulsion System

7.3.2.2 Reaction Control System

The reaction control system of the PLM stage consists of two redundant subsystems configured as shown in Figure 7-29. Each subsystem consists of 8 R-4D thrusters operating on a 1.65 mixture ratio of oxidizer to fuel and fed by two propellant tanks. The thrusters are divided into quadruple clusters which are placed along the periphery of the spacecraft, making a total of 16 thrusters and four propellant tanks for the complete system.

The system utilizes a bipropellant combination of nitrogen tetroxide oxidizer and monometylhydrazine fuel. The propellants are stored in separate spherical tanks of identical size; each tank is 0.76 m in diameter. Both tanks are constructed of a thin steel core overwrapped with prestrssed graphite composite fibers; no thermal insulation material is required. Propellants are equipped with a Teflon diaphragm positive expulsion device which insures eficient tank evacuation.

A pressurant tank stores helium at about 272 atm, and a quad redundant regulator — coupled with a burst disk and relief valve— regulates flow. Together, they insure a 15 atm feed pressure to the propellant tanks, even after any single regulator failure. There are burst disks and pyrotechnically actuated squib valves to isolate propellants from the engine (and high pressure gas from the propellant tanks) until the system is ready for operation. This system also has manual fill and drain valves to load propellant and pressurant gas into the system, as well as additional manual valves for system leak checking on both sides of the pyro-isolation valves and regulators. Check valves insure that the fuel and oxidizer can never mix anywhere in the system, except in the engine. Finally, pressure transducers, filters, temperature sensors, and line and component heaters are provided to ensure proper subsystem operation. A mass distribution of reaction control system components is given in Table 7-12.

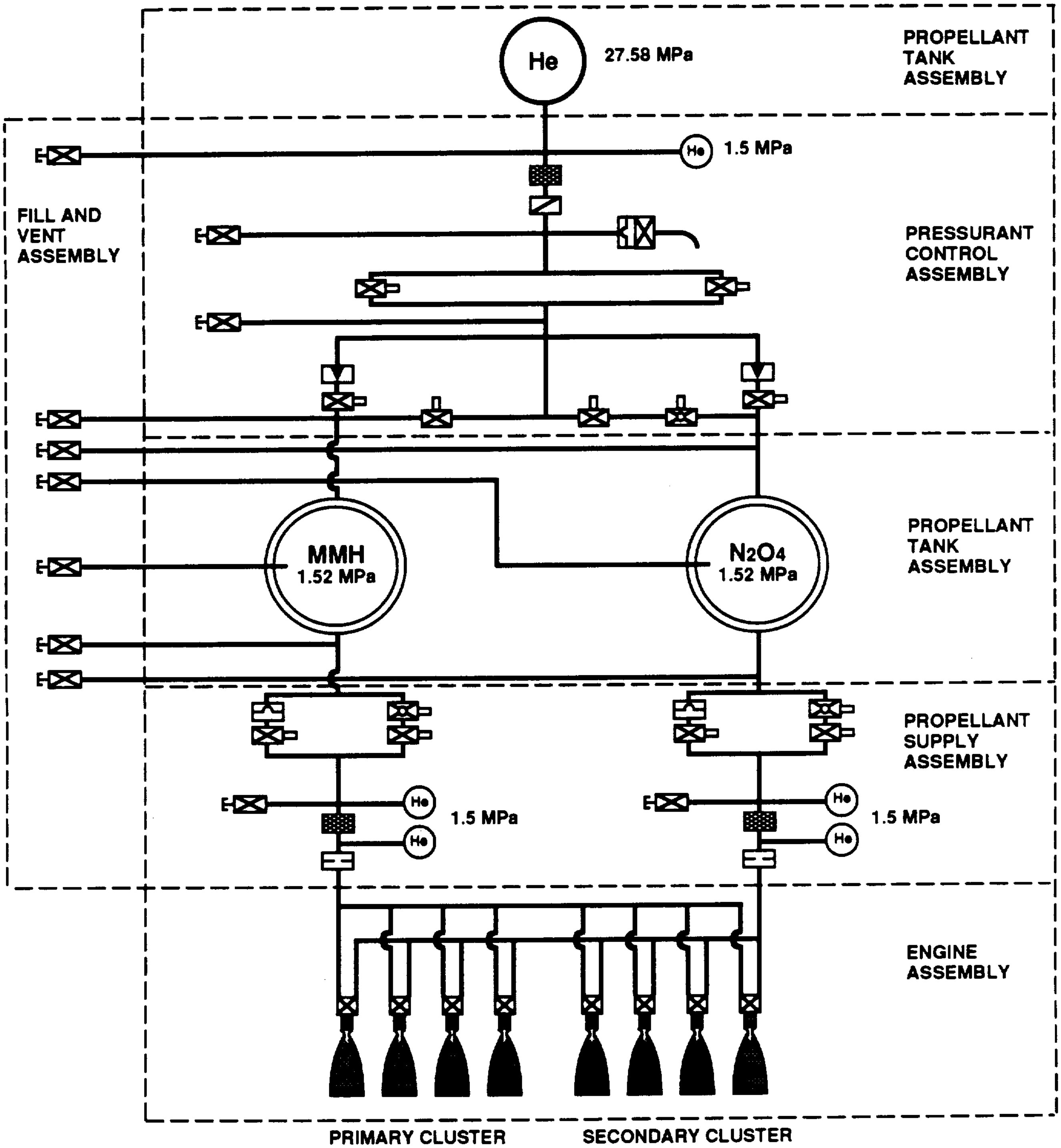


Figure 7-29

PLM Secondary Propulsion System

Table 7-11: Mass Distribution of PLM Primary Propulsion System

COMPONENT	MASS [kg]
Empty Fuel Tank	334
Fuel Mass	549
Empty Oxidizer Tanks	106
Oxidizer Mass	3,032
Empty Helium Tanks	109
Helium Mass	102
Monitoring equipment	20 (estimated)
Propellant lines	26 (estimated)
Valves	39
Engine mass (3 RL10A-4 engines)	504
TOTAL FUELED WEIGHT	4,821 kg

Table 7-12: Mass Distribution of the PLM Secondary Propulsion System

COMPONENT	MASS [kg]
Empty Fuel Tanks	20
Fuel Mass	192
Empty Oxidizer Tanks	20
Oxidizer Mass	317
Empty Helium Tanks	6
Helium Mass	2
Monitoring equipment	20 (estimated)
Propellant lines	26 (estimated)
Valves	62
Engine mass (16 R4-D engines)	60
TOTAL FUELED WEIGHT	725 kg

7.3.2.3 Deployment Engines

The Deployment Package consists of solid rockets for tip-over and liquid rockets for a soft landing.

Three Star 48/TE-M-236 Solid Rocket motors were chosen to tip the precursor lander. The engines are used for SARV retrograde, but also work well for this task. The engines will be tailored down from a rated 7.5 second burn to a 6.5 second burn. This reduction still allows a large margin so that the lander will definitely tip, but it will still have a slow angular velocity. The engines are affixed to the nose of the habitat, pointing away from the top of the primary airlock. The engines are split into a pair and a single so their thrust does not encounter an R4-D engine nozzle. The engines are also angled slightly away from the communications antennae so their exhaust is not detrimental. These considerations diminish performance and add small torques to the system, but these effects are minimal.

The Star48/TE-M-236 motors are 18.3kg each. Tailoring the engines down to a 6.5 second burn time will trim the engine masses down to about 16kg. The motors are 0.324m long, and provide an average thrust of 5600N in vacuum for their burn duration.

Two XLR-132 Liquid Engines provide a controlled descent of the free end of the lander. The computers control the descent using data from a small radar pointed at the ground giving orientation and angular velocity data. Each XLR-132 is 54kg. Each provides up to 16680N of thrust in vacuum. The engine measures 1.2m long and the nozzle expands to a maximum diameter of 0.6m. It achieves an I_{sp} of 340 seconds combining monomethyl hydrazine with nitrogen tetroxide.

7.3.3 Power and Thermal Control

Almost all of the SLURPP system is completely configured and operational at launch. The fuel cells, propellant tanks, Hydrogen dryer, reliquifaction units, water bladder, and power conversion equipment is all configured inside the PLM. To complete setup of the SLURPP system, the astronauts must remove the PV arrays and their associated structure, motors, cabling, etc. and integrate them, Before the main PV arrays are deployed, SLURPP runs off of two auxiliary PV arrays that deploy from the side of the PLM.

7.3.3.1 PLM Power Supply

The PLM is the primary power supply stage of the upper stages of the precursor vehicle. As mentioned before, it will make use of some of the SLURPP fuel cells which it carries for providing power to the PLM stage itself and to the LBM. The PLM was estimated to require 1100 W for 5 days. This power is provided by adding 30.4 kg of reactant mass to the PLM-SLURPP reactant tanks, and by using 15 kg of the SLURPP fuel cells. Furthermore, the PLM must supply power to the LBM. Altogether, for inflight stage power of the PLM and LBM, the tanks of the PLM must hold 51.84 kg of reactants. The SLURPP fuel cells, designed for 35000W, can still easily take care of the inflight power needs. The inflight power reactants break down as 46.07 kg O_2 and 5.8 kg H_2, or as 0.04 m^3 of O_2 and 0.08 m^3 of H_2.

7.3.3.2 PLM Thermal Insulation

Insulation for the PLM stage is designed to allow 0.083% fuel mass boiloff over a period of 30 days.

There are two tanks for both oxidizer and fuel, making a total of four; the radius of each spherical hydrogen tank is 1.535 m, while that of each oxygen tank is 1.101 m. These tanks contain all fuel for this stage, including that necessary for power systems.

Two hundred and twenty-seven layers of aluminized mylar are required to insulate the hydrogen tank, representing a total thickness of 16.29 cm, while the oxygen tank requires only 131 layers totalling 9.40 cm thickness. The total mass of the insulation is 673.34 kilograms for all four tanks.

7.3.4 Guidance and Navigation System

7.3.4.1 Inertial Measurement Unit

The inertial measurement unit, located in the PLM of the precursor mission, will be the same as that discussed in Volume II, Chapter 5. However, because this aspect of the mission needs two levels of redundancy, five of the six gyros and accelerometers may be used. Using five components gives ten possible combinations. The five gyros that are used should be from the list of six component orientations. The IMU should be aligned with the spacecraft coordinates.

7.3.4.2 Star and Sun Sensors

The PLM has four star trackers and one sensor in an arrangement similar to that on the ERM, discussed in Volume III, Section 5.3.4.

7.3.4.3 Data Processing

The data processing of the IMU and sensor outputs should be done onboard during powered flight, such as midcourse corrections and lunar landing. During unpowered flight, i.e. trans-lunar orbit, mission control should monitor the position and orientation of the spacecraft. The process shown and discussed in Volume II, Chapter 5 will be used for attitude and navigation.

7.3.4.4 Radar Altimeters

The PLM has three radar altimeters and one sensor in an arrangement similar to that on the ERM, discussed in Volume III, Section 5.3.4.

7.3.4.5 Antenna Beacons

The PLM has two antenna beacons and one sensor in an arrangement similar to that on the ERM, discussed in Volume III, Section 5.3.4.

7.3.4.6 GPS

The PLM has two GPS systems. These systems are placed near the communicaition equipment. It takes accurate readings to within 4m in LEO every second. These readings are used to update the INS which is the primary mode of navigation. The GPS system is used in both rendezvous operations as well as earth low Earth reentry operatoins.

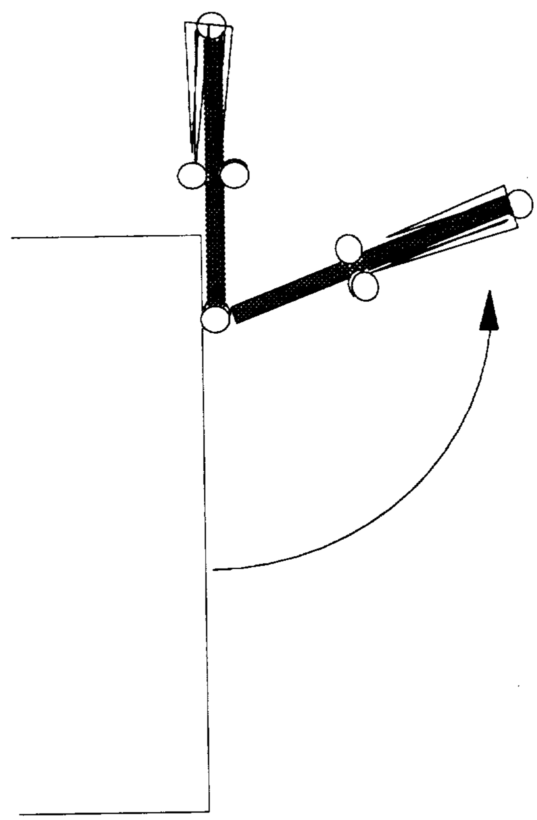
7.3.5 Communications and Control System

7.3.5.1 Communication Antennae

The PLM carries the high gain antennae in its Nose Section. The rest of the communications system such as the computers and data storage resides inside the habitat. These antennae are located on the PLM so that bags of regolith on top of the habitat will not interfere with the pointing system. A detailed discussion of the communications system appears in Volume II sections 4.2.2 and 4.3.4.

7.3.5.2 Communication Antennae Deployment

The communication antennae are folded on the front of the PLM/Habitat. Once the launch nose cone is shedded in orbit, the antennae can deploy. The main boom the antennae are mounted to lies flat on the top of the PLM/Habitat cylinder. After the nose cone is removed, the boom rotates 180° to hang over the edge of the cylinder. Next, the antenna arms fold from flush against the boom to perpendicular to it. Once the structure is open, the antennae open up in an umbrella configuration. These antennae are then oriented properly for communications. Figure 7-31 shows the antennaes' connection to the end of the habitat. Their deployment is shown in Figure 7-30.



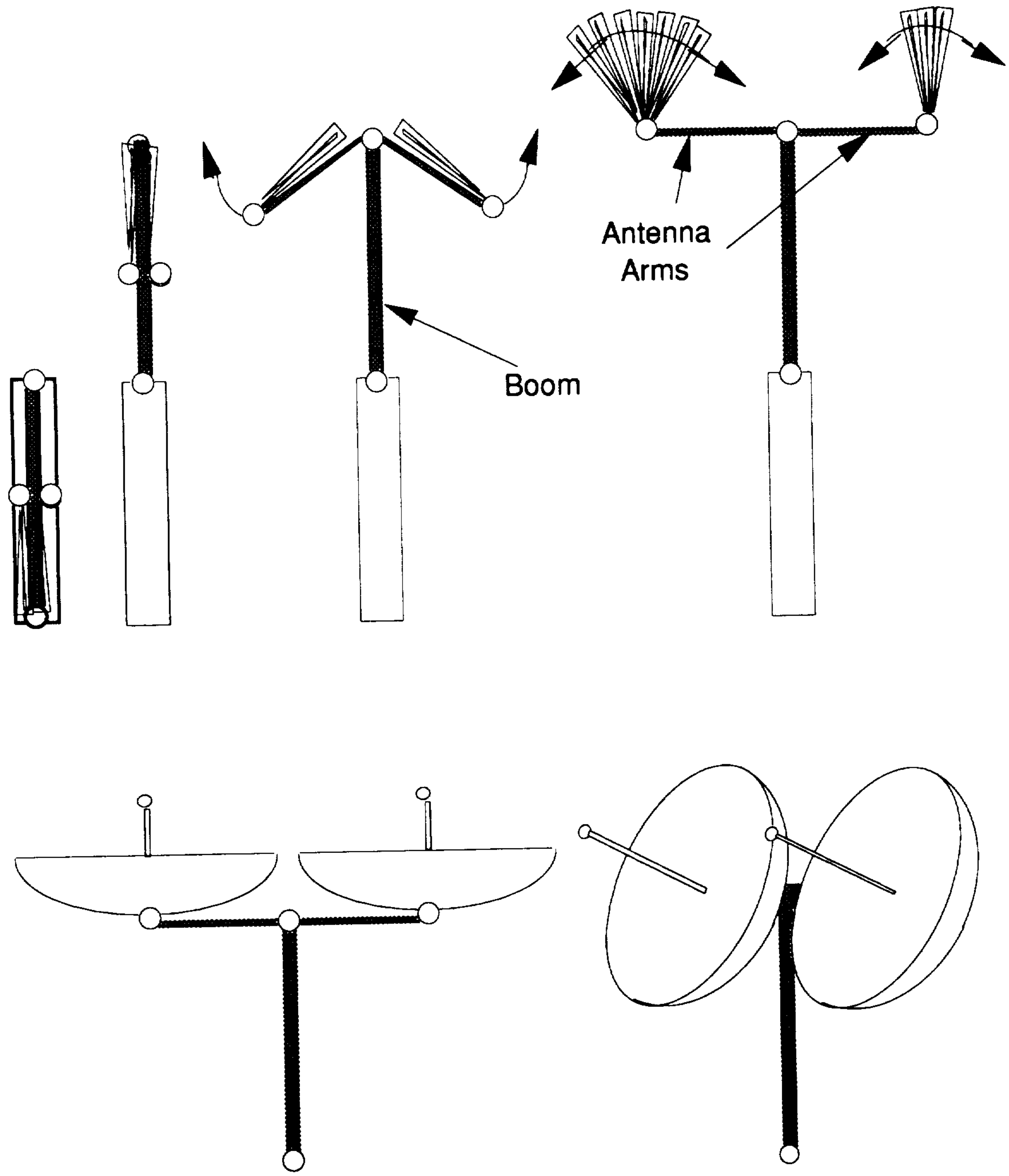


Figure 7-30
Antennae Deployment

Periodically, the spacecraft will alter its orientation with respect to the Earth. At these points, the antennae will rotate to provide the maximum cross-section. One of these times is during deployment. Since the solid tip-over rockets aim in the general direction of the antennae, care must be taken to insure their integrity. The dishes temporarily tilt to avoid the rocket exhaust. Once the precursor lander is horizontal, the antennae reorient toward Earth.

7.3.6 Status Monitoring

7.3.7 Subsystem Interfaces

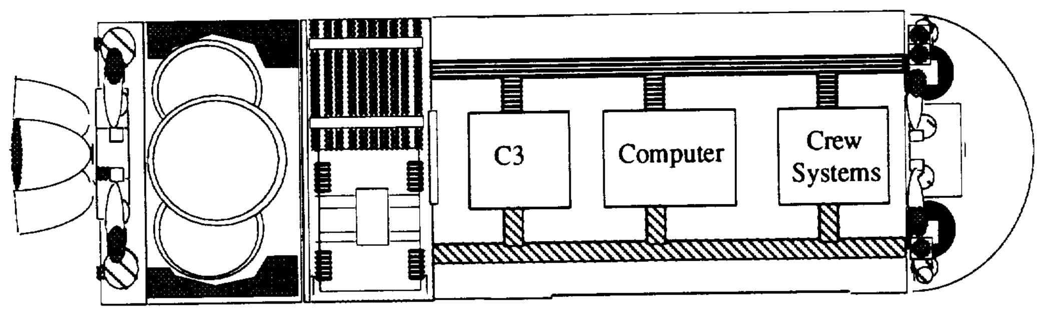


Figure 7-31
PLM and Habitat Interfaces

7.3.7.1 Mechanical Interfaces

The mechanic interfaces consists of airlocks that connect the habitat to the PLM on one end and on the other end provides a exit to the lunar surface.

7.3.7.2 Data Interfaces

The data interfaces consists of fiber optics that transmit and monitor the status of the power subsystem in the PLM to the main computers in the habitat. The data interface also provides the link between the C3 subsystems and crew systems to the computer for monitoring and maintaining the habitat.

7.3.7.3 Power Interfaces

The power interface provides power from the fuel cells in the PLM to all other subsystems.

8. Surface Payloads Description

Surface Payloads include all the hardware delivered to the Moon surface by the Payload Landing Module. The following sections describe the BioCan - the habitat module for the astronauts, the Rover - a multipurpose surface transportation vehicle, the Regolith Collector - a street-sweeper type of vehicle for collecting regolith for protecting the habitat against solar flares, the Conveyer - a segmented conveyer belt used primarily for implementing the radiation protection.

8.1 Habitat Module

8.1.1 Habitat Module Requirements

8.1.1.1 Set-up Requirements

The ECS in the habitat will be fully functional when the piloted mission arrives. The ECS includes the following systems: 1) thermal control, 2) atmosphere supply and control, 3) atmosphere purification, and 4) humidity control. Operation of the ECS in the habitat before the arrival of the crew will allow Mission Control to determine the functionality of the ECS. The communications and control systems in the habitat will also be fully functional before the arrival of the crew.

The crew will arrive on the moon in soft suits. With the ECS in the habitat functioning when the crew arrives, the crew will be able to remove their suits when they get inside the habitat. The remainder of the crew systems in the habitat will require less than eight hours to set-up (three crew members working). This will allow the crew to begin work on the power system and radiation protection soon after arrival.

8.1.1.2 Survivability Requirements

The design life requirement of the habitat module is 12 years. This will allow development of a lunar base during the lifetime of the habitat module. The habitat systems will be modular to allow replacement and upgrading of components.

8.1.1.3 Functional Requirements

The habitat module must provide the following functions: 1) ECLSS (crew systems), 2) communications and control, 3) EVA storage (hard and soft suits), 4) laboratory / system maintenance facility, 5) crew quarters, 6) personal hygiene, 7) galley, 8) dining / recreation / exercise, 9) health maintenance, 10) laundry, 11) circulation (the crew must be able to move about the habitat module in hard suits during an emergency, and 12) protection against moderate solar flare.

8.1.1.4 Abort Requirements

In the event of abort from the habitat, if possible, the habitat should be left in such a way that it can be brought back to functioning ability with as little excess repair payload as posssible. This consideration is mitigated by the time frame within which the abort must take place. Unless the habitat is about to combust, abort from the habitat will not be a split second process. The astronauts face a three day return trip, so any failure without a three day margin (failure not occuring to the habitat, rover or other surface implements), should be dealt with in situ. There are provisions for making repairs for many failures, and these, of course, are not considered abort situations.

If abort occurs because of external dangers, such as increasing radiation levels or developing problems with the crew capsule or ERM, all efforts should be made to follow the standard shut-down procedure for the habitat. If the problem is with the habitat, the offending components should be disabled and if necessary isolated so as to prevent contamination of the rest of the habitat. If possible, the crew should compile a complete checklist of the damage, why it occured, and the necessary parts for repair. This will allow the next crew to begin repair with a minimum of excess material.

In the case of problems with the propulsion systems, it may be necessary to send a rescue mission to the moon. Again, the astronauts would need to survive a minimum of three days without outside support. It is unlikely that there will be another craft ready to launch at just that moment, so a more extended stay may be necessary.

Abort from the habitat is just like shortening the mission. The return of the original capsule and crew cannot occur with a propulsion malfunction. A lunar rescue atttempt by a second craft and crew is the only way to deal with an uncompensated propulsion failure.

8.1.1.5 Modular Requirements

Modularity of the habitat module is important to allow expansion of the lunar base. Additional modules can be connected to the habitat module at the secondary airlock. See 8-1. Passageways between modules would be made using inflatable structures. Using this method of connection, there are always two exit paths from each module.

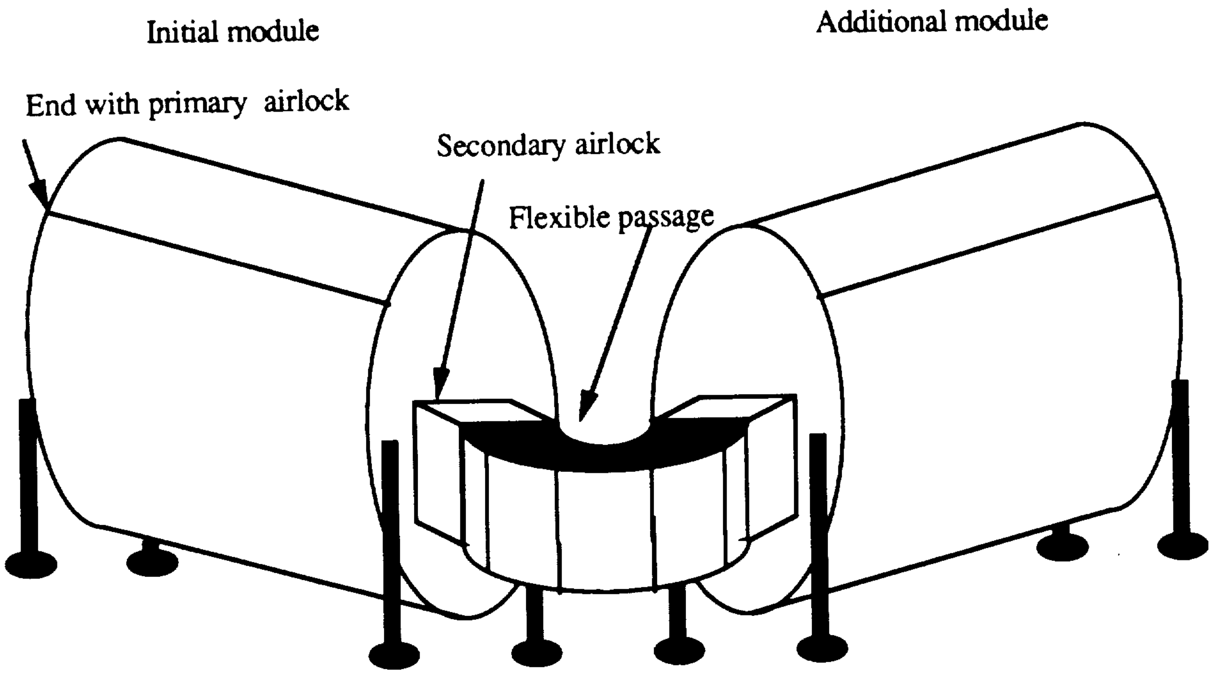


Figure 8-1
Habitat Expansibility

8.1.1.6 Expansibility Requirements (Re-supply)

Recurring missions must carry the additional supplies for the refurbishment of the habitat. Basically, these supplies are the crew provisions (Subsection 8.1.3.1). However, additional tanks must be provided in the crew module in order to carry the extra needed oxygen, nitrogen, and water to the lunar surface. Table 8-1 contains the mass and volume requirements of the refurbishment supplies for the habitat. Thus, the recurring budget for the crew module is the original budget for the crew module (Chapter 6) plus the refurbishment budget given in Table 8-1. This recurring budget is important when considering future missions and the expansibility of the crew module.

Table 8-1: Refurbishment Budget For The Habitat

	Mass (kg)	Volume (m3)
Recurring Totals	**1409.69**	**4.43**
Clothing	-	0.8
Shoes	4	-
Dress (3 weeks)	55.2	-
Food (dry weight)	102	0.83
Oxygen		
Daily Supply	50.93	-
Cabin Atmosphere - three	255.6	-
EVA	70.34	-
Nitrogen		
Daily Supply	100.8	-
Cabin Atmosphere	126	-
Tanks		
Oxygen - three	48	1.59
Nitrogen - three	95.7	0.81
Water - six	109.02	0.34
Water		
Drinking water	282.2	-
Wash water	80.5	-
Water for oxygen reclamation	23.4	-
Tolletries	6	0.06

8.1.2 Structural Design

8.1.2.1 BioCan Lunar Habitat–External Structure

Load Criteria

The structure for the BioCan lunar habitat is expected to endure a 35000 Pa internal atmospheric pressure. It is also expected to endure the axial loads and lateral accelerations of launch on its walls and internal structures. It is not expected to experience the bending stresses present after deployment to the horizontal position, since most of these forces are taken by the PLM Primary Hull . A certain degree of thermal protection is needed in order to protect the habitat from the extremes of the lunar environment.

Configuration

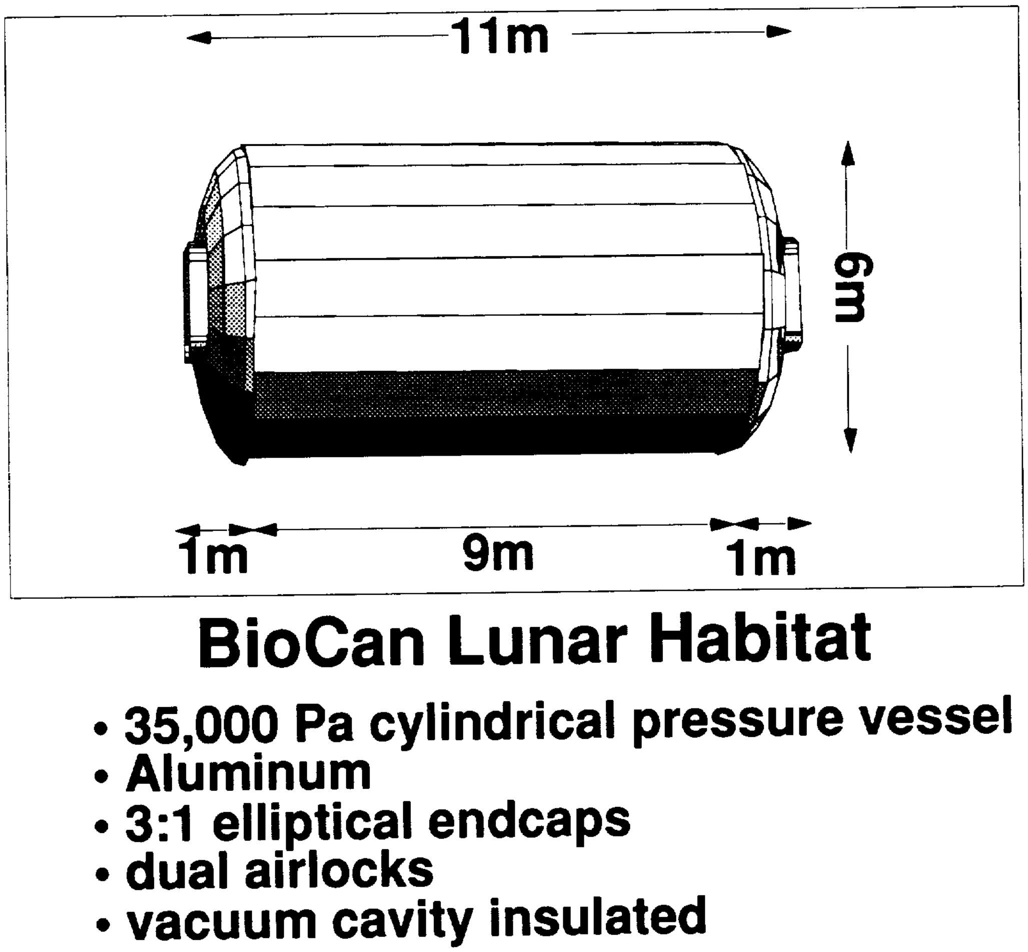


Figure 8-2
BioCan Configuration

A configuration having at least two exit hatches is necessary in case of fire or other emergency. The cylindrical payload area of the launch vehicle puts constraints on the shape and size of the structure. A cylindrical configuration was chosen for the habitat section of the PLM stage (Figure 8-2). The cylinder body has a radius of 2.9 m and attaches to the inside of the frames of the Primary Hull. The BioCan itself is primarily an aluminum pressure vessel with wall thickness of 2mm. The elliptical endcaps have a 3:1 ratio, and extend another meter on each side past the nine meter cylindrical body. From the end of each endcap, the total length of the BioCan is 11 meters. A rectangular airlock exists on each side, situated in the endcaps. Table 8-2 at the end of this section shows the geometry and mass estimate for the BioCan pressure vessel. See Section 7.2.1, *PLM Configuration* for more information about the cargo bay, support legs, and other structures related to the BioCan. Volume II, Section 2.2.5, *Precursor Mission Structures* contains information about the structural analysis of the lunar habitat.

<u>*Summary Specifications*</u>

Table 8-2 : BioCan Geometry and Mass Estimate

GEOMETRY		**MASS ESTIMATE**	
Cylinder Diameter	5.8 m	Material	Aluminum
Cylinder Radius	2.9 m	Mass of Internal Structures	6669 kg
Cylinder Length	9 m	Basic Structure Mass	1245 kg
End Cap Ellipse Ratio	3:1	Airlock & Hatch Allowance	800 kg
End Cap semi-minor axis	1 m	Joints & Fittings	35%
		Total BioCan Structural Mass	**2760 kg**
Total BioCan length	11 m	Total BioCan Mass (full)	9429 kg
Skin Thickness	0.002 m		

<u>8.1.2.2 Internal Structure and Layout</u>

The pressure hull of the habitat is a cylindrical structure with a diameter of 5.8 meters and a length of 11 meters. The pressure hull has elliptical ends with the ratio of the semi-major axis to the semi-minor axis equal to 3:1. See 8-3.

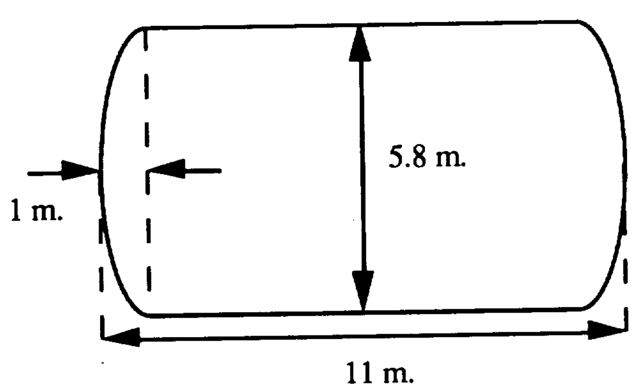


Figure 8-3
Pressure Hull (top view)

The habitat module has two types of floor levels. The crew quarters section of the habitat module consists of two levels. The lower level has a floor to ceiling height of 2.0 meters and the upper level has a floor to ceiling height of 1.3 meters. The width of the floor on the lower level is 5.08 meters, and the width of the floor on the upper level is 5.67 meters. See Figure 8-4.

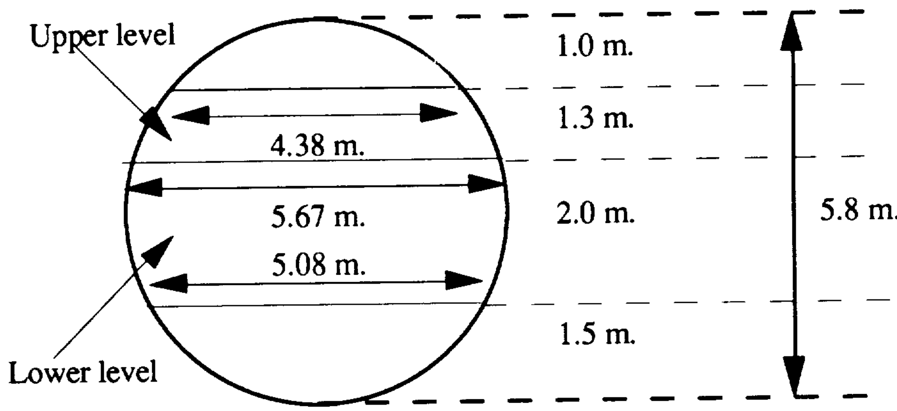


Figure 8-4
Floor Levels of Crew Quarters

The remaining sections of the habitat module utilizes a single level floor. In this area the floor to ceiling height is 2.8 meters. The width of the floor at this level is 5.51 meters. See Figure 8-5.

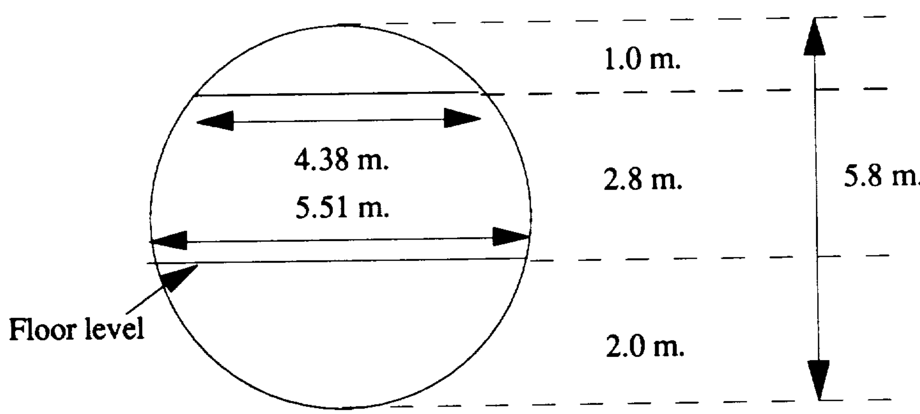


Figure 8-5
Main Habitat Floor Level

The length of the crew quarters section of the habitat module is 3.4 meters. The two floor levels are connected with two .25 meter steps. See Figure 8-6.

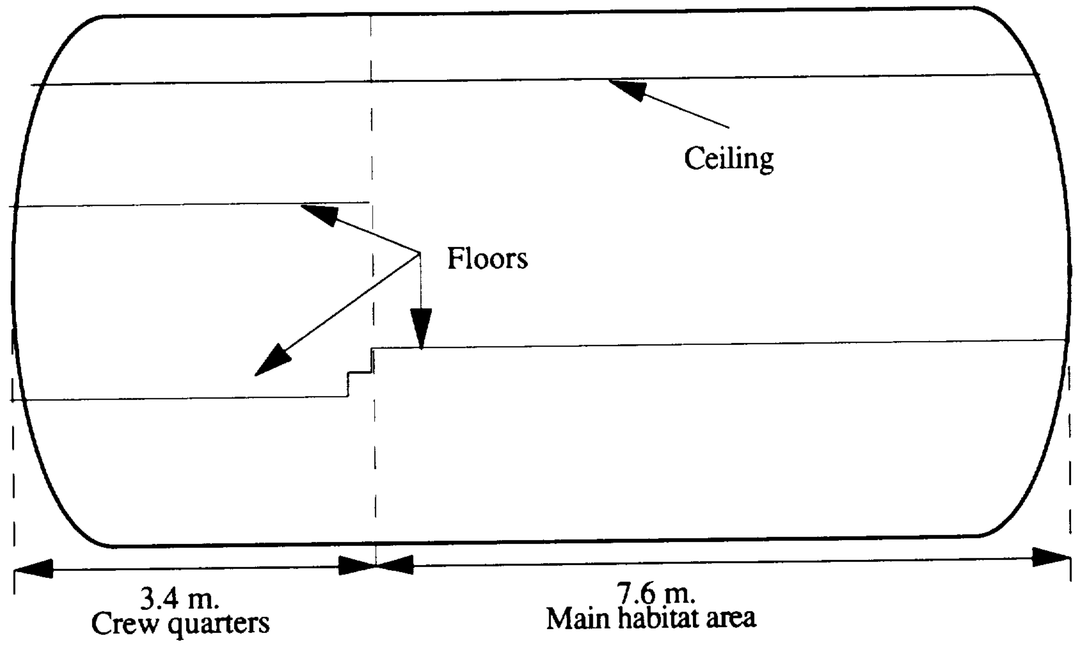


Figure 8-6
Floor Levels of Habitat Module (side view)

The habitat module has an airlock at either end. The circulation paths inside provide adequate space for the crew to move through the module while wearing hard suits. The habitat module is divided into several functional areas. See figure 8-7.

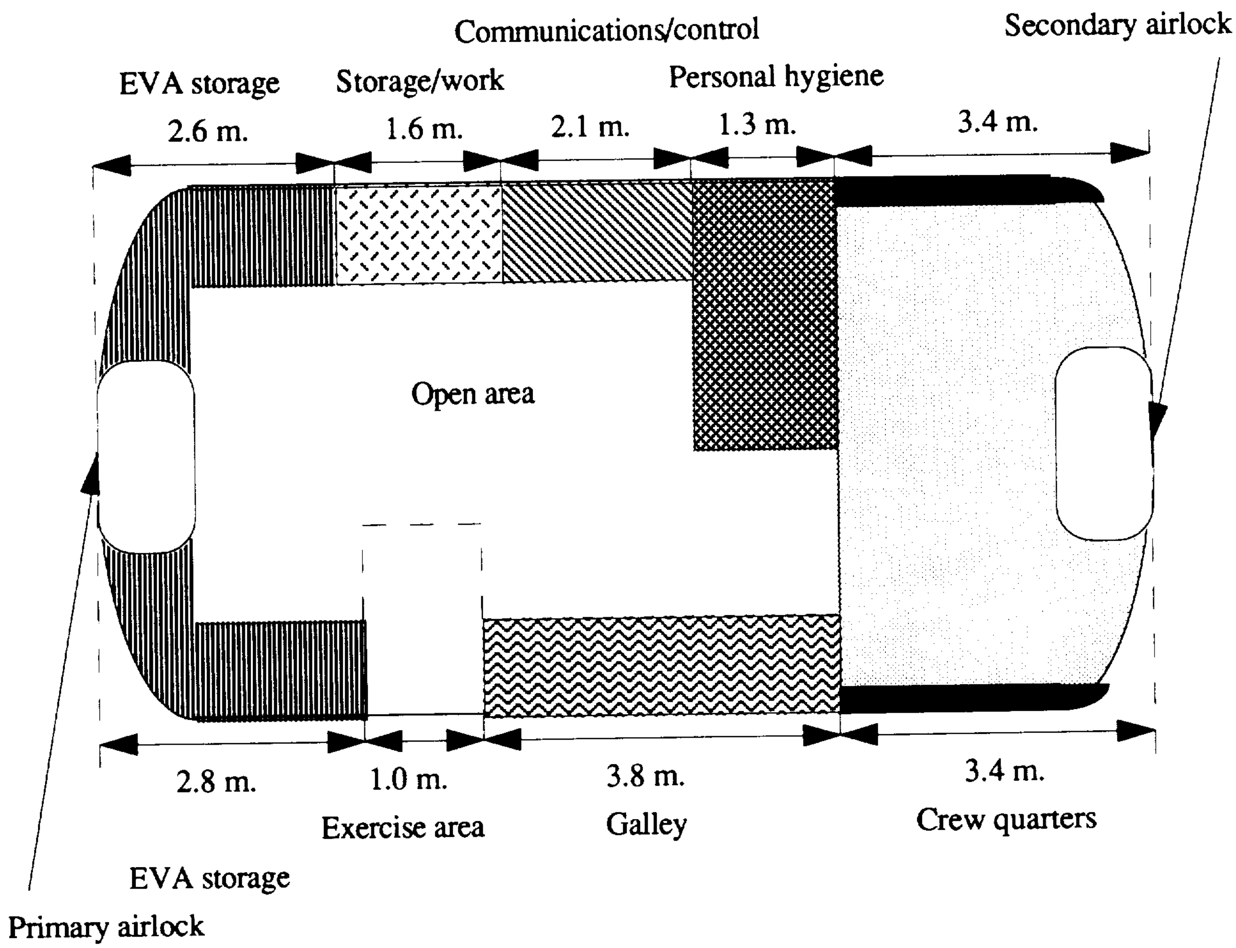


Figure 8-7
Functional Layout of Biocan (top view)

8.1.2.2.1 EVA Storage

The EVA storage area is located next to the primary airlock. It contains sufficient space to store five EVA hard suits and four soft suits. The hard suits will be stored in the habitat module during the pre-piloted mission. The crew will arrive and transfer to the habitat module in their soft suits. The EVA storage area contains a vacuuming system to clean the lunar dust off of the suits.

The hard suits will be hung during storage in the EVA storage area. Each suit requires a volume 2 meter tall, 1meter deep, and .8 meters wide. The four soft suits can be folded into a volume 2 meters high, 1 meter deep, and .6 meters wide. See figure 8-8. There will be hooks in the open area just inside the airlock to hang the hard suits while they are being vacuumed.

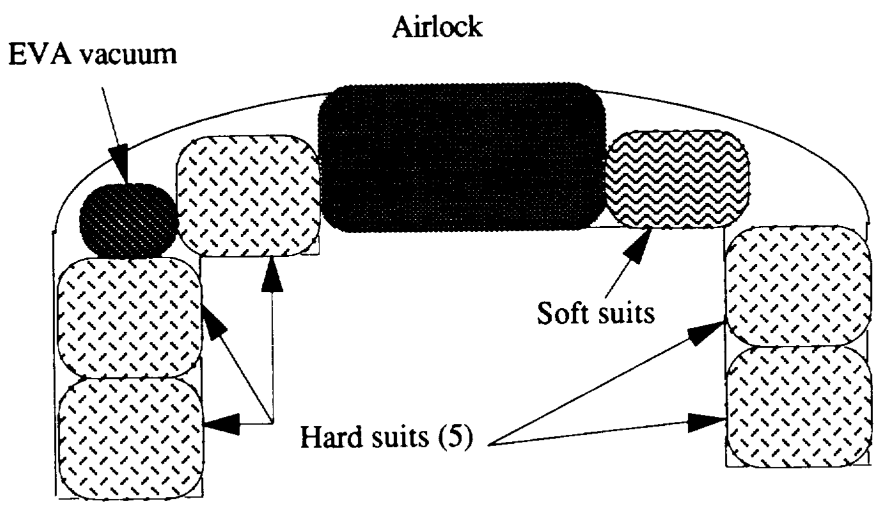


Figure 8-8
EVA Storage

8.1.2.2.2 Storage and Work Area

This area consists of a rack containing any required maintenance and scientific equipment. The rack has a desk area where work can be done. There will be four chairs in the work, galley, and control and communications areas. These chairs can be placed in the area as they are needed.

8.1.2.2.3 Control and Communication

This area will contain all of the communications and control equipment for the habitat module. The communications equipment will include systems to talk with earth and the rover and a intercom system to be used internal to the habitat module. The Control and Communications area will also contain computers for control of the habitat module and for data processing.

8.1.2.2.4 Galley

The galley contains all the necessary systems and storage areas for food preparation. There is space for bulk food storage where food stores are wrapped individually in plastic wrap. The galley also contains ambient storage where plates, utensils, and commonly used cooking items are kept. A microwave oven and dishwasher system which could also be used as a clothes washer will also be installed in the galley. The galley will also contain a water dispenser and deployable counter. Trash will be handled with a trash compactor and a small storage area.

A table 1.5 meters long and .82 meters wide will be stowed in the section of the galley nearest the exercise area. This table will pull out and unfold. The table will be used for both working and eating, and it will be strong enough to support a man so it can be used as an examination table in a medical emergency.

8.1.2.2.5 Personal Hygiene

This will be a closed compartment containing a toilet, sink, shower, and toiletry storage racks.

8.1.2.2.6 Crew Quarters

The crew quarters will be a common four person area. The lower level will provide a small desk for each crew member, storage space for clothing, and a small sitting and reading area. The desk and personal storage space will be located along both walls. Each two person desk will be .5 meter wide and 1.8 meters long. Shelves will be placed above the desk for person storage. The open area of the lower level will be used for reading and relaxing. The dimensions of this area are 1.8 meters by 4.0 meters. This area will provide adequate room for two crew members to do personal work or relax.

The upper level will contain the bedding for the four crew members. The dimensions of the upper level are 5.08 meters wide by 2.6 meters long. The upper level is longer than the lower level because it extends over the top of the airlock into the elliptical endcap of the pressure hull. The upper level will contain four beds that are each 2 meter long and 1 meter wide. A light partition will separate each of the beds on the second level.

8.1.2.2.7 Exercise Area

The main pieces of equipment in the exercise area are a treadmill and an ergometer. The treadmill will be stowed vertically against the pressure hull while it is not in use. The ergometer will be secured to the floor during the flight to the moon. When the crew arrives, the ergometer will be moved to a hanging stowed position just above the exercise area.

8.1.2.2.8 Tank Storage

All of the oxygen, nitrogen, and water tanks will be stored under the floor of the single level section of the habitat module. There is adequate room under the floor of this section to place the following spherical tanks: 1) three oxygen tanks (r = .72 meters), 2) three nitrogen tanks (r = .57 meters), and 3) six water tanks (r = .43 meters). In addition to the tanks, the wash water recovery system would also be placed under the floor.

The humidity control and atmospheric purification systems will be placed above the ceiling. Each of these systems will be attached directly to the pressure hull. All piping for atmospheric supply and control and water management will be run under the floor and above the ceiling.

8.1.2.2.9 EVA Airlock

The following specifications are scaled down from a design for a space station airlock The largest part of the structure is the EVA airlock itself. In the Bio-Can, this structure will occupy approximately 4.2 m3 and will require a mass of about 450 kg. to cover the walls of the structure, equipment attachments, lights, and gas recovery system.

Mass and volume allotments of 25 kg. and 0.028 m3 must be made in order to accomodate the hyperbaric equipment. This apparatus, located under the equipment airlock, allows for the treatment of rapid decompression illness by subjecting the occupant to pressures as high as 5 atm for a period of time. It is these high pressures that require the large mass in wall structure previously mentioned.

The third component of the airlock assembly is the equipment airlock. In the Bio-Can, this will occupy a space of 0.6 m3 and require 82 kg of mass to account for its walls, lighting, and gas recovery system. The structure is provided to mimimize consumables and the time required to pass items between the habitat and EVA crew. It is also used to deliver medicine and food to astronauts undergoing pressure treatment in the EVA airlock.

Both the primary and secondary airlocks are of the same design and dimensions. The total mass requirement for the airlocks is 1100 kg.

8.1.2.3 Overall Specifications

Table 8-3: Mass and Power Budgets

System	Mass (kilograms)	Power (kWH / 24 hr)
Structure		
pressure vessel	2760	-
internal structure	1500	
storage racks	1000	
Crew Systems	3853	
general		92.8
exercise		9.7
commode		.3
housecleaning		.3
airlock		8.0
C^3	316	53.3
Thermal Control	200	24.0
Lighting	90	7.0
Scientific Equipment	250	24.0
total	**9969.0**	**219.4**

Table 8-3 gives the mass and power breakdown for the habitat. The mass figure includes the composite pressure hull for the habitat but it does not include any mass for the outer shell which covers the entire surface payload or the stringers which support the pressure hull. The power figure is given as total power used during a 24 hour period. 219.4 kWH per 24 hours equates to an average continuous power of 9.14 kW. The power system is being designed to support a continuous load of 9.14 kW with peak power at 15 kW.

8.1.3 Crew Systems

The habitat has the primary goal of providing a livable environment for the astronauts. The crew systems requirements include a 99% reliability. This reliability will be achieved by having systems with this 99% reliability already or by providing three levels of redundancy in the systems that do not. [Shea, 1992]. Systems that require redundancy basically have a 95% reliability and when three systems are connected in parallel then the net reliability will be the desired 99%. Crew systems has also established a factor of safety of 1.5 for all consumables. These two aspects, reliability and safety factor, affect crew systems' drivers. The drivers are mass, volume, and power requirements.

Crew systems includes crew provisions, environmental control, and other equipment with regards to mass, volume, and power budgets. The totals for the precursor mission are given in Table 8-4. Each system is broken down completely in this section and further budget elaborations are given.

Table 8-4 : Crew Systems Habitat Total Budget

System	Mass (kg)	Volume (m3)	Power (watts)
Crew Habitat Provisions	1332.89	42.95	0
Habitat Environmental Control	1311.12	11.37	3838.5
Habitat Bioinstrumentation	318.2	2.35	4832
EVA Equipment	413	10.25	0
Other Habitat Equipment	160	1.77	1506.3
TOTALS	**3535.21**	**68.69**	**10176.8**

8.1.3.1 Crew Provisions

The analysis for the required crew provisions for the habitat follows the same methods as the crew module. Refer to Subchapter 7.1 of Volume II for the methods used to obtain the mass, volume, and power budgets. However, there are differences between the crew module and the habitat. The habitat supplies were based on provisions for twenty-eight days with a factor of safety of 1.5. Thus, the supply of clothing, food, oxygen, nitrogen, drinking water, wash water, and toiletries are based on twenty-eight days. Other things to note are the medical kit [Pearson, 1971], additional clothing, and water for oxygen reclamation (Subsection 7.2.1.4 of Volume II). Also, the pressurized volume of the habitat is 200m^3, this is used in determining the mass of cabin oxygen and nitrogen needed (Section 7.7.1 of Volume II).

Table 8-5 provides the consumables for a four person mission for twenty-eight days with a factor of safety of 1.5 built-in. This factor of safety and the three extra cabin atmosphere supplies (in case of depressurization) provide more consumables than required for a four person - twenty-eight day mission if everything goes as planned. The oxygen and nitrogen provide enough for 94.5 days due to the extra supplies in reserve for repressurization atmosphere. However, the drinking water only lasts forty-two days since it only has a factor of safety of 1.5 and no reserve supplies.

Table 8-5: Crew Habitat Provisions

	Mass (kg)	Volume (m3)
Crew Provisions total	**1332.89**	**42.95**
Crew of four	-	40
Clothing	-	0.8
Shoes	4	-
Dress (3 weeks)	55.2	-
Sleepers	32	1.2
Food (dry weight)	102	0.83
Medical kit	42	0.06
Oxygen		
Daily Supply	152.85	-
Cabin Atmosphere - three	255.6	-
EVA	70.34	-
Nitrogen		
Daily Supply	100.8	-
Cabin Atmosphere	126	-
Drinking water	282.2	-
Wash water	80.5	-
Water for oxygen reclamation	23.4	-
Toiletries	6	0.06

8.1.3.2 Environmental Control

The habitat crew environment is engineered to provide the most comfortable conditions for the astronauts. The environmental factors include atmosphere, water, and waste. The habitat will utilize a semi-regenerative system as described in Subchapter 7.2 of Volume II. The system is regenerative in oxygen (Subsection 7.2.1.4 of Volume II) and wash water recycling (Subsecton 7.2.2.1 of Volume II).

Figure 8-9 is a diagram of the habitat's environmental control and waste management system. Table 8-6 contains the total budgets for the system. This system is based on the trade and selection analysis given in Subchapter 7.2 of Volume II and is described in full in the following sections.

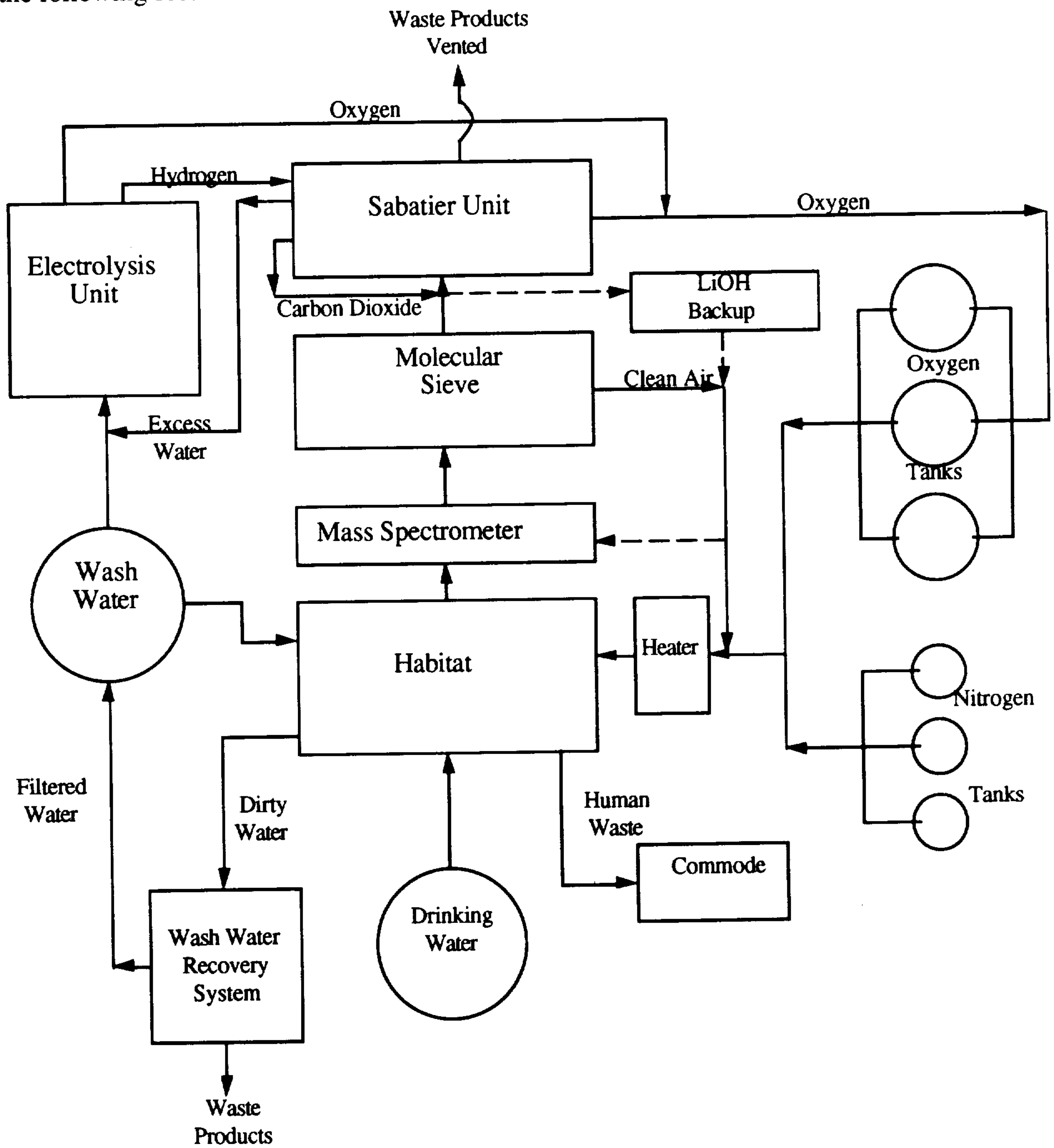


Figure 8-9

Habitat Environmental Control and Waste Management System

Table 8-6 : Habitat Environmental Control Total Budget

Environmental Control Total	Mass (kg)	Volume (m3)	Power (watts)
	1311.12	**11.37**	**3838.5**
Tanks			
Oxygen - three	48	1.59	-
Nitrogen - three	95.7	0.81	-
Water - six	109.02	0.34	-
Waste Management			
Commode	46	0.24	340
Water Management			
Humidity control	115	0.76	725.2
Wash water recovery system - 3	68.1	0.18	240.9
Tanks	36.4	0.11	-
Piping, etc.	25	2	-
Atmosphere Purification			
LiOH system	60	0.7	80
Molecular sieve - 2	284	0.85	900
Thermal Control System	109	1.03	1000.7
Atmosphere Supply + Control			
Mass spectrometer	18.2	0.1	100
Breadboard 2-gas control	22.7	0.2	100
Lockheed electrolysis	129.5	0.28	287.5
Sabatier/Toxin Burner	77.3	0.28	10.2
Tubing, etc.	30	1.8	-
Fire Suppression and Detection	37.2	0.1	54

The power levels given in Table 8-6 are just the required level for each component. The total given is just a sum of these levels. Subsection 8.1.3.2.4 contains a power profile for crew systems' part of the habitat.

8.1.3.2.1 Atmosphere

The general composition of the habitat atmosphere is identical to the atmosphere of the crew module. Section 7.2.1 of Volume II contains the engineering of the atmosphere and the reasons for choosing the following characteristics.

Total Pressure = 0.34 atm

Nominal Partial Pressures =

Oxygen = 0.218 atm

Nitrogen = 0.122 atm

Carbon Dioxide < 0.0102 atn

Water Vapor = 0.0082 atm to 0.0184 atm

Temperature = 17.8° to 27.2 ° C

Mixture (by volume) = 64% oxygen and 36% nitrogen

Table 8-6 shows the other atmosphere necessities. For atmospheric purification, a molecular sieve will be used to remove carbon dioxide as described in the oxygen reclamation section (Subsection 7.2.1.4 of Volume II). For redundancy, an extra molecular sieve and a LiOH system {(adsorbs carbon dioxide from the air), Subsection 6.3.2.1 of Volume II} will be provided for the habitat. The habitat also includes a thermal control system, atmospheric supply and control equipment, humidity control system, and fire suppression and detection equipment. Table 8-6 contains all the mass, volume, and power budgets for these systems [Pearson, 1971] and [Shewfelt, 1992].

8.1.3.2.2 Water

The habitat will recycle wash water (Subsection 7.2.2.1 of Volume II). This is shown in Figure 8-9. The recycling of the wash water provides a mass savings of 75.6 kg in the second mission while costing only an additional 28.9 kg in the first mission (Subsection 7.2.2.1 of Volume II).

8.1.3.2.3 Waste

The Columbiad lunar habitat will include one Allied-Signal commode unit for the disposal of human waste, wipes, and potentially other soft disposable items (see Vol. II section 7.2.3 for details). In addition to this unit, the habitat will have a central garbage storage bin for such materials as food packaging remmants, uneaten food, used personal hygiene items (dental floss, tissue paper, etc.), spillage containment bags, and any other garbage gathered. The bin will enable crewmembers to manually crank a piston in order to compact accumulated. The compacted trash will be occasionally be placed in sealable bags for storage or lunar surface burial. Garbage storage areas will be connected to an air circulation conduit and filter/freshener to eliminate cabin odor. Several waste baskets will also be supplied to place at various locations about the habitat for temporary, convenient non-toxic trash disposal.

As on the Crew Module, the there will be an extensive effort placed on minimizing disposable food packaging in the lunar habitat. This packaging accumulation is a much more critical factor on the habitat where the astronauts will be consuming three meals per day for 28 days. Freeze dried foods will be wrapped in celophane and eaten on reusable, multi-compartment plastic trays. The celophane is very compactable and will contribute very little to garbage volume. Beverage powders will be stored in large, permanent cylinders with turnspouts to eliminate packaging, and cups will be reusable.

The waste management equipment will also include a handheld vaccum capable of intaking small liquid and solid spills. Vacuum containment bags will be highly resistant to volatile contents to prevent releakage. Hence, full bags can be placed inside the central garbage storage bin for storage or burial.

The Crew Module Environmental Control System includes an air filter system to reduce atmosphere particulate count to healthy levels. The system also includes a Mass Spectrometer which will be able to detect trace levels of predetermined expected toxins which will be periodically monitored by crew and mission control.

8.1.3.2.4 Power

Table 8-6 provided the power levels for the various components of the environmental control system for the habitat. However, these are just values and do not provide a power profile. All of these systems run continuously except for the commode and the EVA hard suit recharge system . The total power that these systems require is 3868.5 watts. The commode runs an average of 14 times per day for the four-person crew. The commode requires 340 watts of power for a duration of 20 seconds each time it is in operation [Shewfelt]. The EVA hard suit recharge system (Section 7.7 of Volume II) requires 500 watts for a period of ten minutes four times a day. Another aspect of crew systems is bioinstrumentation and exercise equipment (Subsection 8.1.3.4). In terms of power, the bioinstrumentation and exercise equipment require 4832 watts of power for a two hour period during a typical day. This allows for thirty minutes of exercise for each astronaut. A final aspect of crew systems is housekeeping (Subsection 8.1.3.5). This requires a power level of 556.3 watts for 30 minutes each day. Figure 8-10 shows crew systems daily power profile for the habitat. The commode's power is shown as spikes, the EVA charge's power as slightly wider spikes, the housekeeping's power as a solid jump, and the exercise period's power as a huge increase. The profile is in terms of a 24-hour period, however, "hour zero" is not necessarily equal to 12:00 a.m. The figure provides a typical power profile with the required continuous power and the additional spikes in power along with their durations.

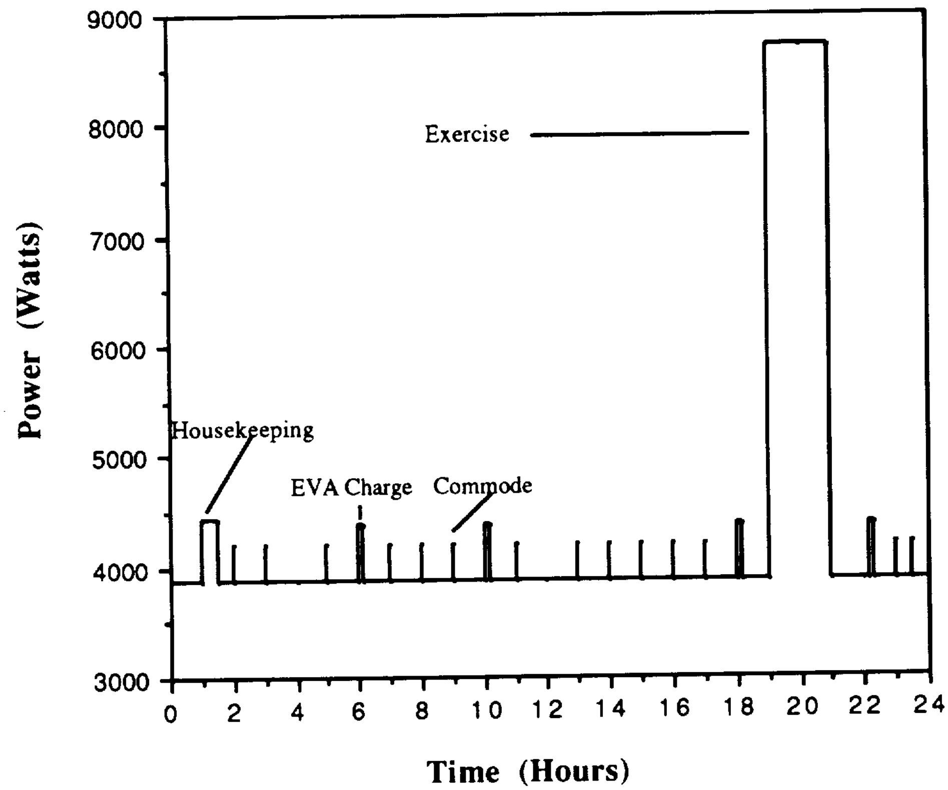


Figure 8-10
Crew Systems Daily Power Profile For The Habitat

8.1.3.2.5 Fire Detection and Suppression

Fire is a grave danger is space. Possible causes are overheating of electronic equipment and astronaut error. The system implemented in the habitat is very similar to the Space Shuttle's current system and the proposed system for Space Station Freedom. The technology is based on work done at AiResearch [Shewfelt, 1992]. The habitat contains twenty-eight smoke detectors, five fire detectors, and twelve fire extinguishers (five of which is built into the system, seven of which are portable). The system mass is 37.2 kg and volume is 0.1 m^3 (Table 8-6).

8.1.3.3 Crew Garments and EVA Suit

8.1.3.3.1 Shirtsleeve and Undergarments

Columbiad crewmembers shall wear a variety of undergarments to remain comfortable both within the EVA spacesuit as well as during non-critical phases of lunar habitat occupation. This includes a set of Capellene underwear to provide warmth and a layer against outer suit friction irritation. For long term EVA suit wear, the crew will also don a Fecal Collection System similar to those worn on Apollo missions. During habitat occupation, crewmembers will don Shuttle multi-pocketed pants and flight jackets which provide comfort, warmth, and are highly functional, as well as lightweight tennis shoes.

8.1.3.3.2 EVA Suit

The spacesuit to be included on Columbiad for lunar EVA will be based on the suits currently being researched by NASA for use in Space Station EVA. At Ames Research Center at Moffett Field, California, two suits are under scrutiny. These are the AX-5 (see Figure 8-11) and Mark 3. Each suit will provides the necessary protection against the thermal, chemical, and radiation environments of space. In addition, both incorporate a high level of protection against micrometeoroid impact, a likely event during long periods of Space Station construction in LEO. This armor-like quality is equally necessary on the lunar surface not only for micrometeoroids and flying impact debris, but also will resist

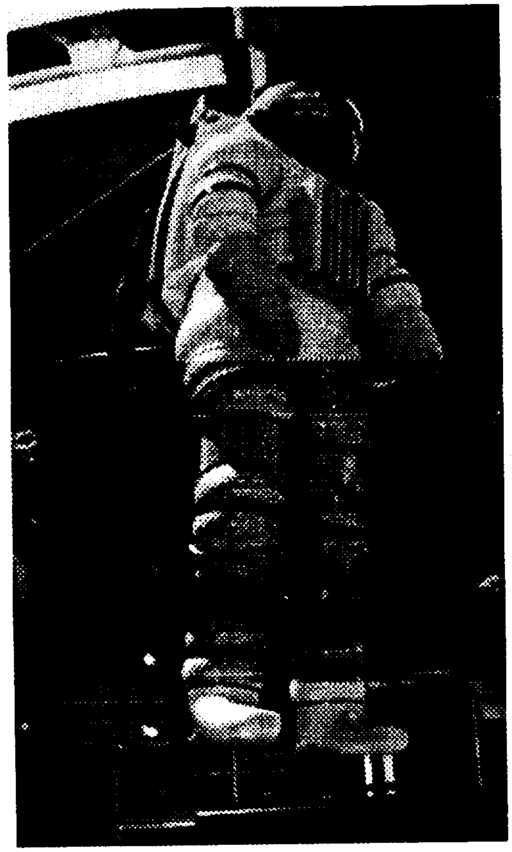


Figure 8-11
Ames AX-5 Hardsuit

tearing and puncture over the many hours of EVA expected over the mission. What is more, both suits have been designed with constant volume joints to significantly reduce resistance of motion over Apollo suits. With such ease of motion, the astronauts may be able to work up to eight hours per excursion, increasing mission productivity. Both suits are capable of operating under 0.56 atm as opposed to the 0.29 atm of the current Space Shuttle suits. Though all crew inhabited volumes will include an atmosphere of only 0.34 atm, this extra pressure capability will eliminate pre-breathing procedures before EVA. This factor is crucial to a mission which may include frequent egress and ingress each working day. The decision has been made to include a suit for each crewmember plus a spare suit to replace an irreparably damaged suit.

Though both suits have been designed for the same task, the AX-5 and Mark 3 vary considerably. The AX-5 is an entirely solid design, comprised of a double hull aluminum and stainless steel body. Having an entirely solid design makes suit fabrication much more predictable and reliable than fabric suits. Furthermore, the AX-5 has only 15 major parts, making maintenance simple. Finally, differently proportioned astronauts are accomodated through the use of sizing rings at various joints. The Mark 3 combines hard and soft components which enhances flexibility. The torso is of an aluminum construction, and includes a joint at the waist which allows the wearer to bend forward. This section also includes sizing rings for personalized fit. The legs, arms, and boots of the Mark 3 are of fabric construction. Both designs include helmets much larger than current Space Shuttle helmets, allowing for much greater visibility, are of rear entry configuration, and incorporate integral portable life support systems (PLSS).

The exact design for the Columbiad lunar EVA suits will most likely include a combination of features from the AX-5 and Mark 3 suits, plus will have features specifically designed for the Moon rather than for Space Station construction. For example, the joints in the Ames suits have been configured under the assumption that the astronaut would be floating in space, rather than standing in a 1/6 Earth gravity environment. The AX-5 forces the wearer's legs apart for a comfortable floating position. This position is disadvantageous for extensive walking. The current model AX-5 does not include a waist joint, which is a necessary element for lunar applications. Though the suit's PLSS is permanently attached to the suit, Columbiad requires that the integral backpack be alternately supplied by umbilical. During rover transportation, it is desirable to supply the astronauts with a life support system stowed permanently on the rover. This will allow for maximum EVA endurance at the worksite and enables the astronauts to feed off of a reserve supply on the rover if the rover malfunctions. Another issue which must be addressed is that of the effect of lunar dust on the solid joints of the Ames suits. Presumably these joints can be protected against intrusion of particles and resulting joint degradation or failure. Finally, the suit should include hardpoints for lunar tool stowage. Despite the obvious differences between the existing suits and the probable final design, the Ames prototypes best represent the most modern in space suit technology. Additionally, the Space Station suit research is sufficiently far along as to give a much higher probability for meeting project deadlines.

The EVA PLSS unit supplies the spacesuit with a 100% oxygen supply at 0.34 atm. It will provide 8 hours of nominal oxygen supply as well as 1/2 hour worth of emergency oxygen. Exhaled air is processed through a miniature molecular sieve to separate carbon dioxide from breathable air. Carbon dioxide is radiated to space. In addition, the PLSS is integrated with a liquid cooling garment to provide a comfortable temperature for the working astronaut. The working fluid is transported to the pack, and sublimated, releasing the heat to a radiator and outer space. Finally, the EVA PLSS supplies an amount of drinking water to a straw located in the helmet for astronaut consumption during EVA. A battery pack will supply all systems with power 12 hours (reflecting a FOS of 1.5).

8.1.3.4 Bioinstrumentation

Biomedical monitoring of crew members in the lunar habitat and during surface EVA will be performed to assure crew safety and to acquire more extensive data on the physiological effects of microgravity (one-sixth-g).

Exercise Capacity and Metabolic Analysis. Biomedical monitoring will be performed during daily exercise on a treadmill and bicycle ergometer throughout the 28-day lunar habitation. Data on oxygen consumption, carbon dioxide production, lung volume and respiratory exchange will be collected with a metabolic analyzer and studied to detect decreases in exercise capacity. (see Volume II: Section 7.6.1)

Exercise Equipment. Resistive forces encountered in the EVA spacesuit during surface activities necessitates the use of equipment for maintaining musculoskeletal strength and endurance. Trunk and leg muscle atrophy will be decreased by walking or running on an angled treadmill under gravitational loading for at least 30 minutes per day (see Volume II: Section 7.6.2). MK-I exercisers transported from the capsule will be used on the lunar habitat to maintain arm strength (see Volume II: Section 7.5.2).

Medical Kit. In order to accommodate the longer duration of lunar habitation, the first aid kit supplied in Biocan has a larger supply of medication than the kit on the crew capsule. (see Volume II: Section 7.6.3) In addition, blood and urine analysis chemistries will be provided in this kit to chemically detect chronic musculoskeletal atrophy.

Table 8-7 summarizes the bioinstrumentation which will be supplied on the lunar habitat.

Biobelt Assembly. The biobelt assembly underneath the EVA suit provides a critical link between the astronaut and Mission Control during lunar surface activites. Monitors within this assembly provide indicators of potential health hazards such as overheating and overwork. The biobelt assembly is composed of three identically sized signal conditioners (5.84 cm x 3.81 cm x 1.04 cm) which derive power from a self-contained 10 V DC to DC Converter. The *Electrocardiogram Signal Conditioner* develops a signal wave ranging between 0 and 5 volts peak-to-peak which is representative of a crew member's ECG activity. The *Impedance Pneumograph Signal Conditioner* develops signals corresponding to respiration rate over a wide dynamic range of respiratory activity. Concurrently, the *Body Temperature System* outputs a voltage in the range of 0 to 5 VDC corresponding to sensed temperatures of 303 to 319 K (see Volume II: Subsection 7.3.2.2.2)

Table 8-7: Bioinstrumentation on the Lunar Habitat

Parameter	Mark I Exerciser	ECG System	Treadmill	Ergometer	Metabolic Analyzer	First Aid Kit
Number Supplied on Habitat	(2 from Capsule)	1	1	1	1	1
Dimensions						
Height (m)	0.20	0.11	0.14	1.12	1.00	0.64
Depth (m)	0.52	0.39	0.82	0.50	0.55	0.22
Length (m)	0.20	0.46	2.10	2.85	0.79	0.40
Volume (m^3)	0.02	0.02	0.24	1.60	0.43	0.06
Mass Per Item (kg)	5.49	8.60	163.60	50.00	90.00	6.00
Power Per Item (W)	-----	100	4160	72	500	-----
Cost Per Item ($)	75.00	5000	5000	3600	15000	150.00
Supplier	NASA	Siemens-Burdick	Siemens-Burdick	Siemens-Burdick	Wilkin Collins	Zee Medical

8.1.3.5 Other Equipment

Section 7.7 of Volume II provided trades and selection on the necessary additional crew system equipment for the habitat. Table 8-8 contains the budgets for this additional equipment.

Table 8-8: Crew System Additional Equipment For The Habitat

	Mass (kg)	Volume (m3)	Power (watts)
Other Equipment TOTAL	**160**	**1.77**	**1506.3**
Hardsuit Recharge System	50	1.5	500
Lighting	10	0.1	450
Tools, cleaning equipment	100	0.17	
Housekeeping			556.3

8.1.4 Radiation Protection

8.1.4.1 Regolith Support Structure

8.1.4.1.1 Side Ramps

Load Criteria

The Regolith Support Structure is base framework for the lunar regolith, or dirt, which covers the BioCan lunar habitat and protects it from radiation. The density of lunar regolith is approximately 1200 kg/m3, and an 80 cm layer is to be deposited. In addition, the load of any machinery that must climb onto the shield during the construction process must be taken into account. A conveyor belt machine has been chosen as the primary construction vehicle, and its load on the shield is assumed not to exceed 800 kg/m2 in the following calculations. The structure is also loaded by its own weight, which for this case, turns out to be minimal compared to the other loads.

Configuration

The latitude of the landing site dictates the path of the sun as seen by the lunar habitat during the 28-day stay. The BioCan will be positioned with the ends of the cylinder pointing perpendicular to the morning sun. At an equatorial landing site, the sun would pass directly overhead, and the radiation shield would only need to cover the sides of the

BioCan cylinder. However, at higher latitudes, the Sun's arc is inclined, and some protection must be afforded the end of the BioCan facing the Sun at high noon.

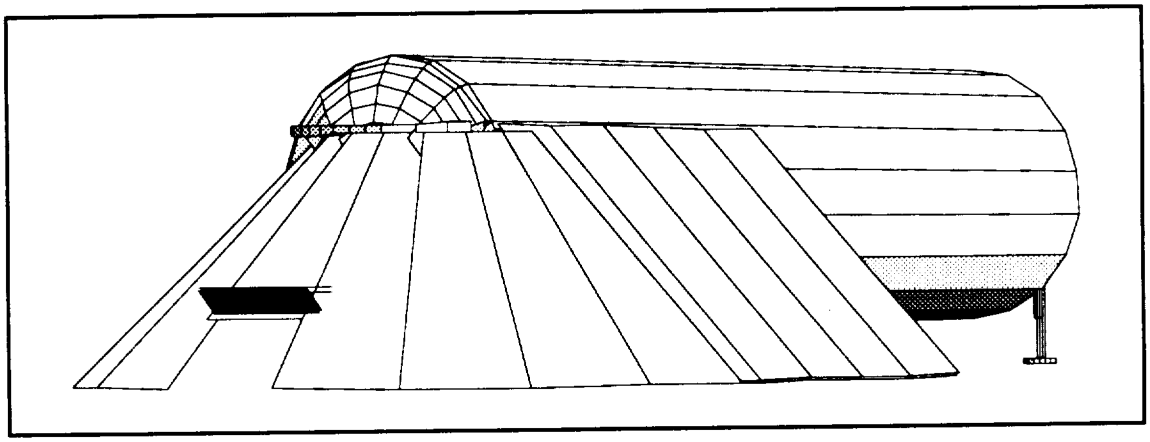


Figure 8-12
Regolith Support Structure Configuration

A configuration was chosen which consists of side ramps which lean against the side of the habitat, and a conical-shaped canopy with covers the end of the BioCan (Figure 8-12). This arrangement provides excellent coverage for most latitudes. A landing site at the pole, however, would see a sun which travels all the 360° around the habitat, making total coverage necessary. For this case, additional shielding would need to be designed which is not included in this report.

Easy setup and compact packaging were also desirable, so a configuration was chosen which consists of many smaller sections which are disassembled and stacked in the cargo bay during the journey. Assembly and installation will take place upon arrival of the crew. Each panel section is assembled as shown in Figure 8-12. There are 5 sections in each of the two side ramps, and 9 sections in the canopy. Each of these sections consists of three plates stacked end to end vertically. A hollow cylindrical beam fits through a slot on each vertical side of the plates, such that two of these beams connect all three plates together. Crossbars are built into the plates and help support the skin laterally. A locking mechanism is present of the side of each plate. This lock is engaged after the beam is inserted into the plate slot to insure that it does not slip in the slot.

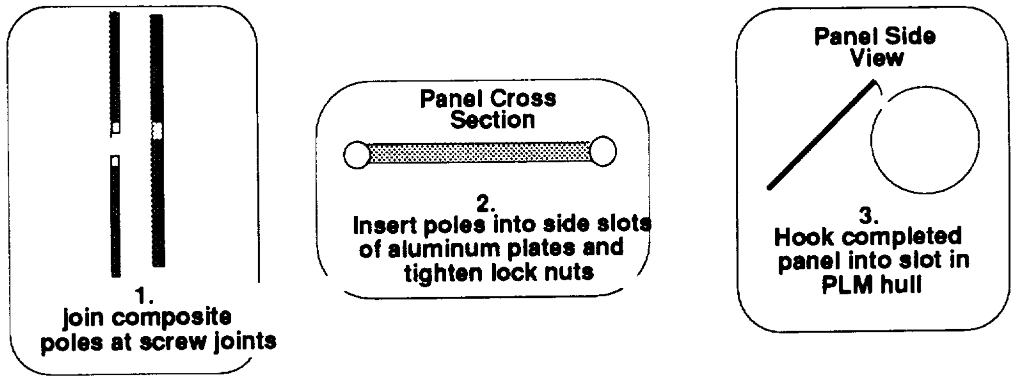


Figure 8-13
Regolith Support Panel Assembly

Each panel section of the Regolith Support Structure is basically a skin suspended on a beam frame. At design time, specifications were not available on the material properties of cloth mesh materials, such as those of graphite or nylon fibers. Aluminum was chosen as the skin material in this design. However, a design utilizing a mesh skin would probably result in weight savings over the aluminum skin implementation. The poles are constructed of graphite/epoxy composite I0I HTS.

8.1.4.1.2 Canopy

The canopy section of the Regolith Support Structure is very similar in construction to the side ramps. All sections are designed with the same cross-section for beams and thickness for skin, but the dimensions are slightly different to account for the curving attachment surface on the elliptical end cap. The method of attachment is the same for the canopy panels as the side panels—an attachment bar runs along the face of the endcap, and the hooked ends of the panels slide in from the top. The middle section of the canopy has open space for an accessway. Above this accessway is a small lip which keeps the regolith above from sliding down over the opening.

8.1.4.1.3 Summary Specifications

Table 8-9. Regolith Support Structure Geometry and Mass Estimates

MASS ESTIMATE		**GEOMETRY**	
Single Support Beam mass	19.9 kg	**ramp angle**	**45°**
Crossmember mass	4.7 kg	height of ramp	4.9 m
beam mass per panel	58.5 kg	**length of ramp**	**6.90 m**

Beam subtotal	1111.7 kg	number of sections per side	5.00
		number of sections in canopy	9.00
Skin mass per panel	88.7 kg	**total number of sections**	**19.00**
skin subtotal	1684 kg		
		number of beams per section	2
		number of crossmembers per section	4
		panel length	2.30 m
regolith support subtotal	2796.2 kg	panel width	2.00 m
Joints & Fittings	10%	panel thickness	0.0035 m
		beam type	cylindrical
regolith support structure mass	**3075.8 kg**	beam outer radius	0.030 m
mass per panel	161.9 kg	beam inner radius	0.020 m

8.1.5 Thermal Control

The issues involved with the habitat thermal control deal with the instrumentation cooling in the biocan and maintaining a comfortable and steady atmosphere temperature in the biocan when the astronauts arrive. The temperature requirements for the biocan are different when it is uninhabited because the instruments can be stored at lower temperatures.

8.1.5.1 Instrumentation Cooling

The location of the instruments in the crew capsule and in the biocan allows the use of cold plates to dispose of excess heat. Most of the instruments are secured on shelves with free space above and beneath the instruments. Cold plates secured to the bottom of the instruments could be installed into the shelves. This would remove the excess heat from the instruments and transfer it to the cabin atmosphere where the cabin atmospheric conditioning unit could remove the excess heat from the cabin. The material chosen for the cold plates is aluminum with wax for the phase-change material.

8.1.5.2 Biocan Cooling

Prior to astronaut arrival there will be no thermal insulation on the exterior of the spacecraft except for the protective paint on the outer skin. This means that 10% of the solar radiation will be absorbed into the outer wall of the spacecraft. Because of the pressure vessel design of the biocan walls, very little of the solar neat will reach the inside of the biocan. An air-conditioning system was chosen to control the internal temperature of the biocan.

After the arrival of the astronauts, the exterior surface of the spacecraft will be covered by bags of lunar soil. The internal heat sources for the biocan will be the excess heat from the instruments and the excess heat from the fuel cells. The biocan temperature range requirements while it is vacant is -55°F. While the biocan is occupied, the temperature needs to stay between 60°F and 80°F.

The design of the air-conditioning system is very basic. Air is vented from the biocan at the end with the emergency exit through pipes where it will be passed through two blowers. From the blowers the air is passed through to a heat exchanger where the air can either be heated or cooled depending on the information that will be fed back from the thermostats positioned inside the biocan to the heat exchanger. Once the air passes through the heat exchanger it is sent back to the biocan where it is split by a vent screen and blown back in. The heat exchanger will use water for the working fluid and will require one pump to move the water around the cycle. The mass estimates for this system are given in Table 8-10 and Figure 8-14 shows a diagram of the air-conditioning system.

Table 8-10 : Mass Estimates for BioCan Thermal Control Equipment

Component	Mass (kg)
Pipe	9.6
2 Blowers	10
1 Pump	10
Total	29.6

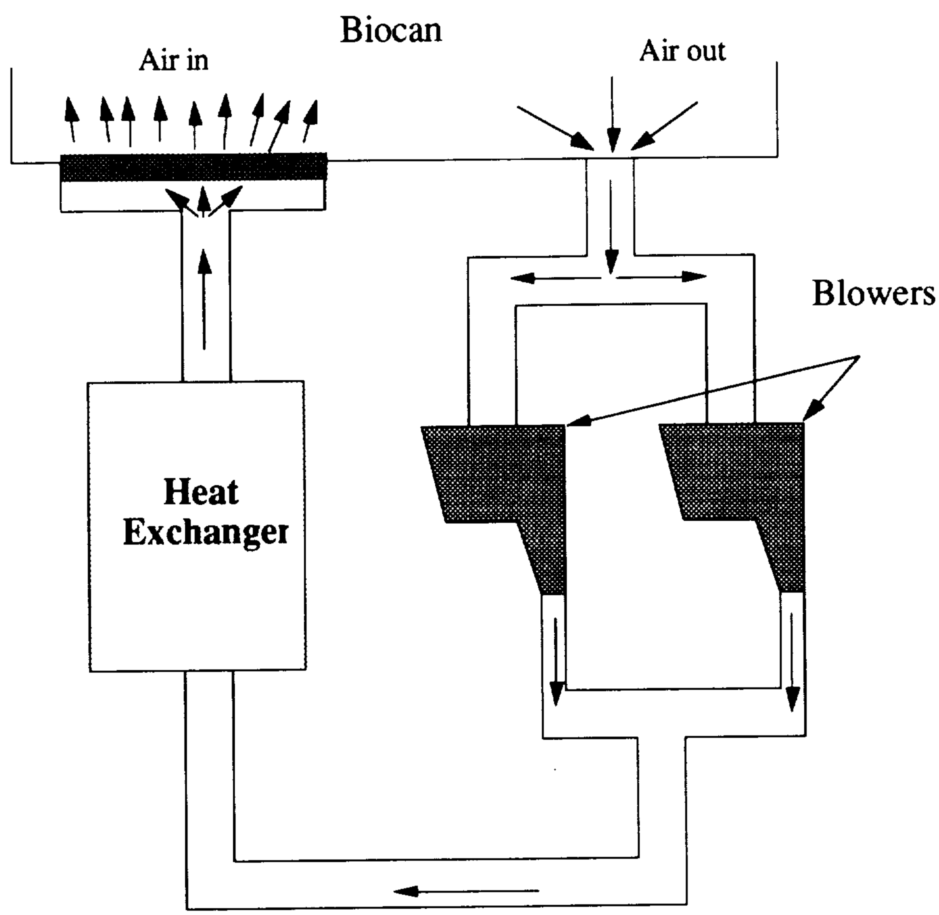


Figure 8-14
Diagram of Air-Conditioning System

8.1.6 Lunar Base Command, Control and Communications

The communications system on the habitat includes a direct link with the Earth, with the Crew Module and with the lunar rover. This system is described in detail in Volume II sections 4.2.2 and 4.3.4.

8.1.7 Status of Habitat

The habitat has two important phases of monitoring. The first is before the astronauts arrive. During that time, it is necessary to do a complete checkout of the habitat and its systems to ensure that all is in order for the astronauts to arrive. If substantial damage has been done to the habitat during transport, it may be necessary to alter the mission, by shortening the stay on the moon. It will definitely be necessary for the astronaunts to send back data on the damage so it can be avoided in the future.The second phases in monitoring while the astronauts are in situ. During this time, damage prediction and prevention are the critical factors. Detecting leaks and repairable damage can substatially increase the life of the habitat.

8.1.7.1 Pre-Piloted Check-Out

During the mission and after deployment of the habitat, it is necessary to stay up to date with the status of the habitat. The most urgent piece of information to be determined is the integrity of the habitat structure and various tanks. After that, the communications and GNC systems will be examined. After the intial checkout, some systems will be continually monitoring and sending back to earth.

Continuously Monitored Systems

- cabin pressure
- tank pressure
- cabin temperature
- power levels
- incident radiation
- composition of atmosphere

These will be monitored inflight and throughout the mission.

The next class of monitoring deals directly with what happens after impact on the lunar surface. The deployment of the habitat will occur before the astronauts arrive. This process has potential to damage the habitat. Using strain gauges on the legs, mercury levels, and contact pads, the habitat will be leveled on the lunar surface and an complete system checkout will occur. (See previous structures sections to see mechanisms for deployment and leveling.)

Deployment Monitoring

- integrity of habitat by checking pressure and strain gauges on legs
- communication system will take up contact with earth
- atmosphere system will be flushed
- GNC systems will be examined

If the system passes to the satisfaction of ground and crew, the manned mission will commence. If not, there may be modifications made to the mission.

8.1.7.2 Monitoring during Habitation

During habitation, the purpose of status monitoring to avoid potentially disasterous situation and to give the crew sufficeint warning time if disaster is imminent. Many of the environmental systems are not redundant, because they are not life threatening. However if a problem developes with, for example, waste disposal, it must be solved. A complete breakdown of what is in the habitat can be found in the Appendix. Particularly critical for monitoring during habitation is the status of the atmosphere. A leak of either the habitat or the tanks can substantially shorten the mission. The monitoring of the habitat must pay special attention to the suits and to the potential loss of atmosphere through any airlock connections. The habitat has a more complex environmental system than the crew capsule and there are more repair facilities available to the astronauts. In general however, monitoring the habitat is similar to monitoring the crew capsule.

8.2 Lunar Surface Power Plant Design

The power requirements for all operations on the Lunar surface comes from the following sources :

Table 8-11: Power Requirements Breakdown for Surface Payloads

Sources	Reqd. Power	Duration
Habitat (incl. scientific expts.)	12 kW	Average continuous
Rover	7.5 kW	8 hours/24hour cycle
Regolith Collector	7.5 kW	8 hours/24 hour cycle
Lunar Conveyer	6.5 kW	8 hour/24 hour cycle
Outdoor Lighting	5 kW	Continuous during Lunar Night

The total stored energy for night power requirement is 11, 250 kWh. The maximum average daytime power requirement is 30 kW.

8.2.1 Solar Lunar Power Plant (SLURPP) Design Requirements

With the above mentioned requirements as a baseline, the following parameters were used in designing the Solar Lunar Power Plant :

- •Lunar Daytime Usable Power Output of 35000W continuous
- •Lunar Nighttime Power Output of 35000W continuous
- •Total System Mass of approximately 10000 kg max

•Can be assembled by a maximum of two astronauts

8.2.2 Solar Lunar Power Plant (SLURPP) System Description

8.2.2.1 SLURPP Overview

SLURPP will provide continuous usable power of 35000W through lunar day and night. Night power will be supplied by a LOX-LH2 fuel cell system which outputs water as a product of the cell reaction. This water will be stored and converted back to LH2 and LOX during the following day through electrolysis. Power for the electrolysis units will be provided by the solar array which will be sized to generate 35000W of usable power for surface operations during the day in addition to the power needed for the electrolytic recharge of the night power system. The habitat will use about 15000W during the day, leaving a surplus of 20000W for recharge of the lunar rover's power system as well as other apparatus. Thus SLURPP will be a continuously self-sustaining solar power system through lunar day and night. (See Figure 8-15.)

Solar Lunar Power Plant (SLurPP)

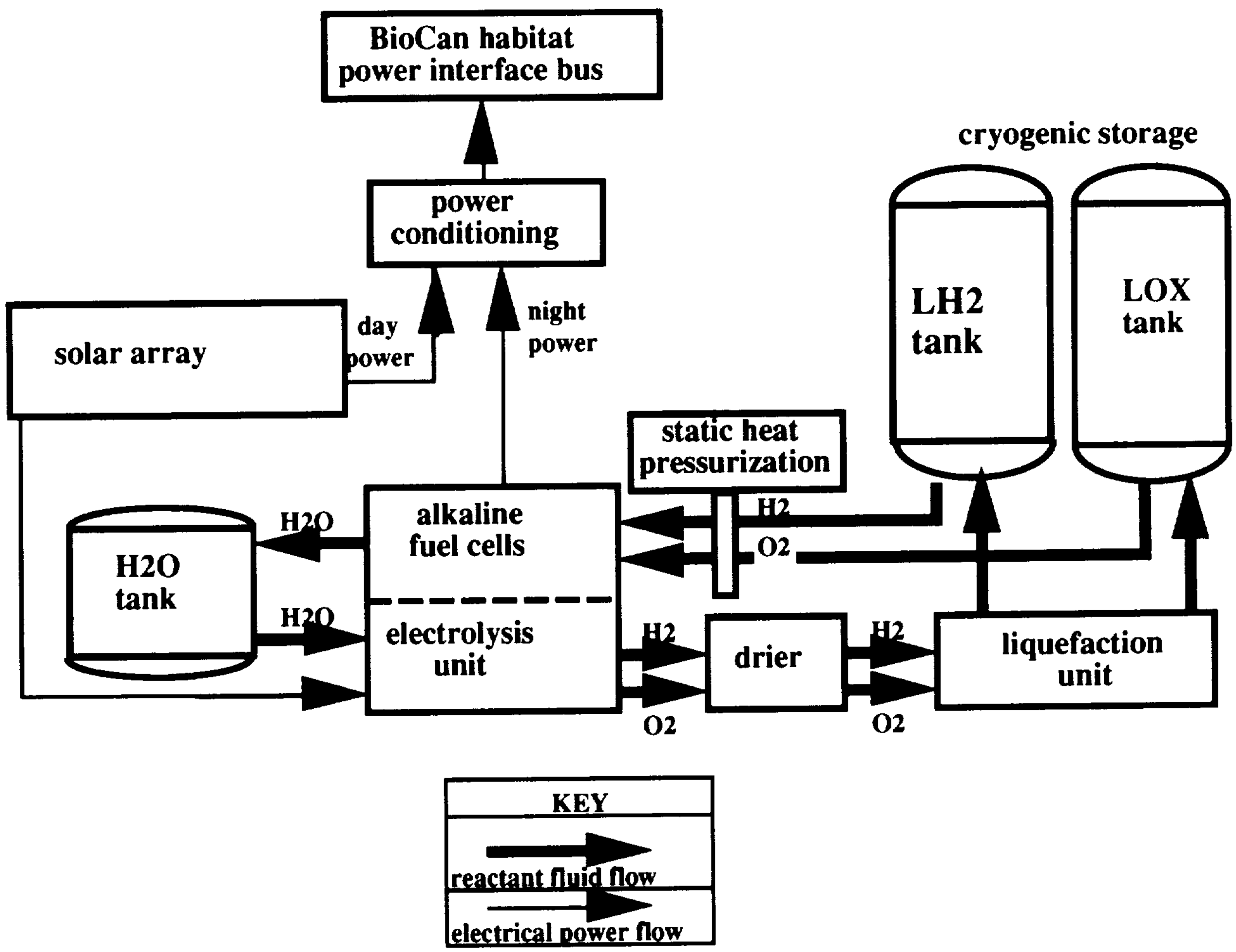


Figure 8-15 : Schematic View of Solar Lunar Power Plant

8.2.2.2 Solar Panel Design and Description - Lunar Daytime Power Generation

8.2.2.2.1 Solar Cells and Circuit Design

Solar Cell Selection

The solar cell selection criteria specified a solar cell with a high conversion efficiency, high open circuit voltage capability and a high spectral response for a large bandwidth. These properties would insure high performance levels from the cells and would maximize the output of the solar arrays. Temperature coefficients were another consideration. These coefficients give the optimal performance temperature and indicate the performance level drop for incremental changes in operating temperature. Because the lunar surface temperature experiences a 288K change between lunar night and day, this performance level drop needed to be investigated.

Solar cells from Astropower, Spectrolab, Inc. and Boeing were considered for use in the arrays. Their properties are given in Table 8-12. Most of the cells show an open circuit voltage drop of approximately -2mV/°C. The lunar day surface temperature is 388K. The efficiency of the cells was calculated for room temperature, or 28°C. This translates to 301K. The actual V_{oc} for the cells, if the cells were to reach an equilibrium temperature of 388K, would be 174mV less than what is listed. This represents the worst case scenario for the performance of the cells.

Table 8-12 : Types of Solar Cells

Company	Solar Cell	Efficiency	V_{oc} (mV)	Temperature Coefficient (mV/°C)
Astropower	AP-102-104	15.7%	540 to 600	–
Boeing	GaAs/GaSb	30.8%	1,000	–
Spectrolab, Inc.	Silicon K7700A	15.4%	600	-2
	GaAs/Ge	18.3%	1,020	-1.8
	GaAs/GaAs	21.5%	1,080	-1.8

The masses of the actual solar cells are minimal, therefore, it is not necessary to consider their relative masses for the final selection of the solar cell. The temperature coefficients do not vary considerably for any of the different cells. The only solar cell that shows any advantage to the rest is the GaAs/GaSb tandem cell from Boeing. This is the cell that was chosen for the solar arrays.

GaAs/GaSb Solar Cell

The GaAs/GaSb solar cell by Boeing is a tandem solar cell. This means that it uses two different dopant materials in the gallium matrix to increase the efficiency of the cell over all.

The dopant elements, Arsenide and Subdenum, respond to different wavelengths of the spectrum. Using both elements increases the efficiency of the cell because the gallium-subdinum cell absorbs part of the energy that the top-layer, gallium-arsenide cell cannot absorb. This increases the overall bandwidth that the tandem cell can respond to and convert into electricity.

The gallium-arsenide cell is stacked on top of the gallium-subdenum cell. The gallium-arsenide cell does not absorb well in the infrared range of the spectrum. In order to allow the energy in the infrared range to pass through to the gallium-subdinum cell located beneath the gallium-arsenide cell, the traditional solid metal contact backing for the gallium-arsenide cell has been replaced by a fine wire mesh. Spreaders are used to dissipate the heat away from the solar cells to the ceramic heat spreaders. The electrical connection is made on a ceramic wiring card located between the two cells. The stacking sequence for the tandem solar cells is shown in Figure 8-16.

The 1V open circuit voltage is achieved using a triplet wiring scheme. Separately, the gallium-arsenide solar cell has a maximum voltage capability of only approximately 370mV. In order to match the voltage capability of the gallium-arsenide solar cell, a triplet wiring scheme is used. The triplet wiring scheme uses three of the GaAs/GaSb tandem solar cells arranged such that the three top gallium-arsenide cells are wired in parallel and the three lower, gallium-subdinum cells are wired in series. This wiring scheme is shown in Figure 8-17.

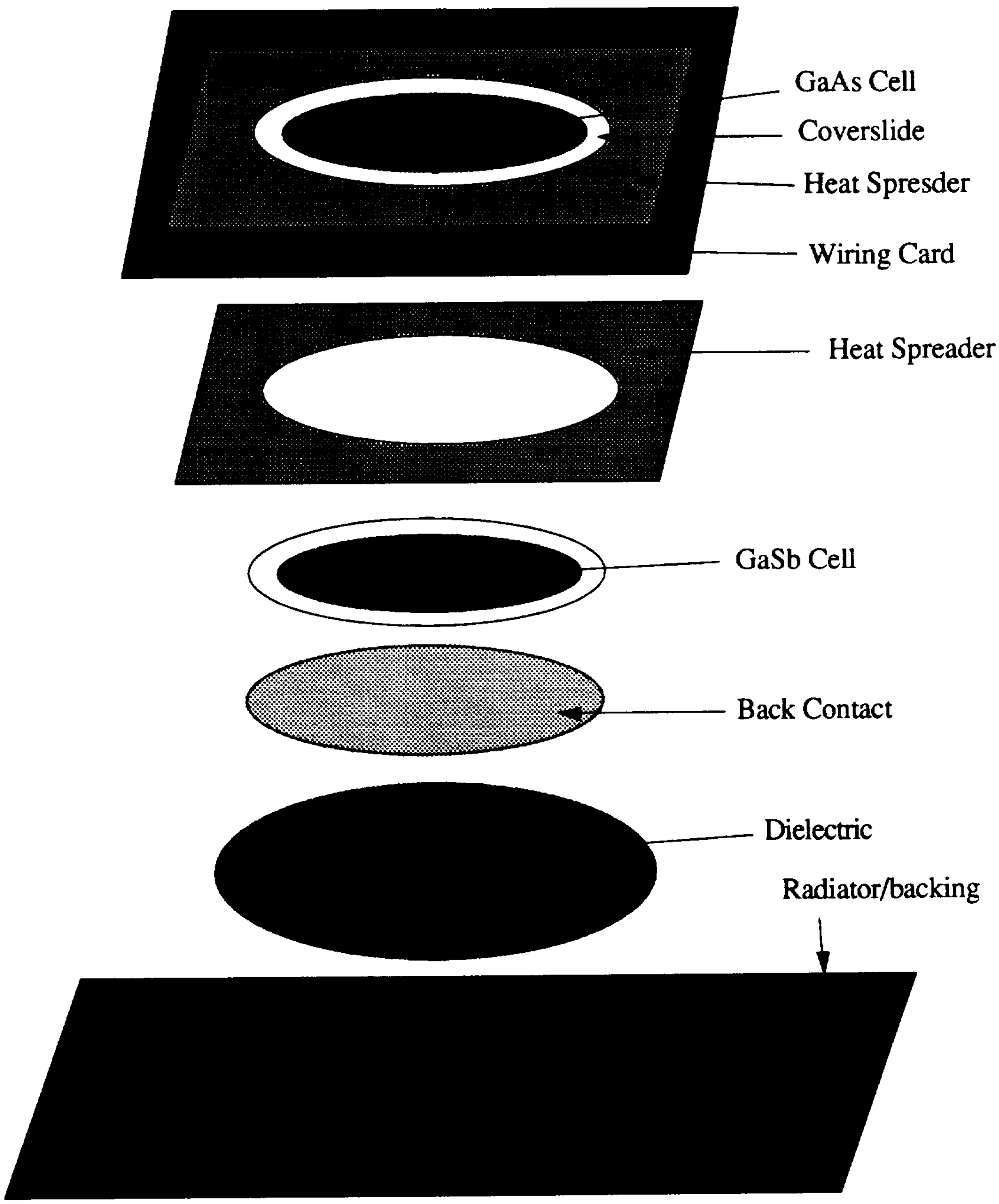


Figure 8-16
Stacking Sequence for Tandem Cells

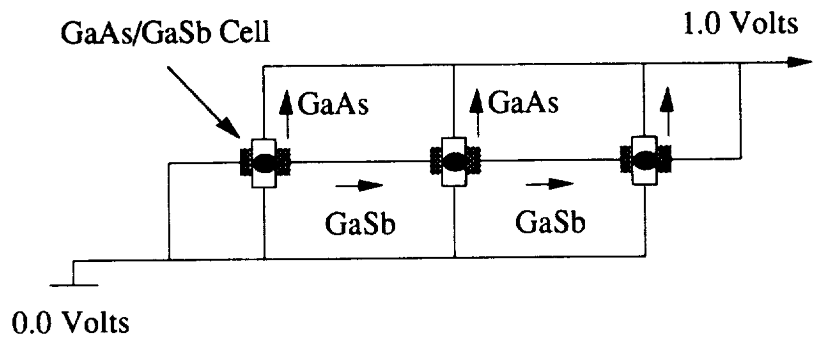


Figure 8-17
Wiring Scheme for Matched Triplet

For each triplet the voltage should remain at 1V even if two of the top gallium-arsenide cells are not functioning, given that the remaining gallium-arsenide cell has not been damaged. The three gallium-subdenum cells are arranged such that if any one of the cells were to not function, the remaining two cells would still be connected to produce a voltage of 740mV. Similarly, if two of the cells were to quit, the resultant voltage would be 370mV from the lower layer of cells. This wiring scheme makes it difficult for an entire triplet to malfunction. Even though the damaging of one cell would cause a decrease in the efficiency of the triplet, the entire triplet would not cease generating power.

8.2.2.2.2 Solar Array Design

Two different array designs were considered for use with the Solar Lunar Power Plant. The first design that was looked at used Fresnel lenses to concentrate the incoming sunlight and focus it onto the solar cells. The Fresnel lenses refract any light which passes parallel to the solar cell that strikes the curved potion of the lens and redirects it onto the active section of the solar cell. The Fresnel lenses would also provide protection for the cells from solar proton flares [U. Washington, 1990]. The lenses would be supported by a honeycomb housing structure. The diagram for the support structure and the lenses is shown in Figure 8-18

Table 8-13 : Solar Array Estimates - Design 1

Array Component	Material	Thickness (mm)	Mass/Area (kg/m^3)	% Total Mass
Lens	glass	0.15	0.49	20

Lens Prisms	silicon	0.15	0.19	7.8
Honeycomb	aluminum	0.15	0.91	37.3
Cell (assembled)	–	0.46	0.05	2.0
Radiator	aluminum	0.20	0.55	22.5
Radiator Coating	alumina	0.01	0.08	3.3
Miscellaneous	–	–	0.17	7.0
Total			2.44	

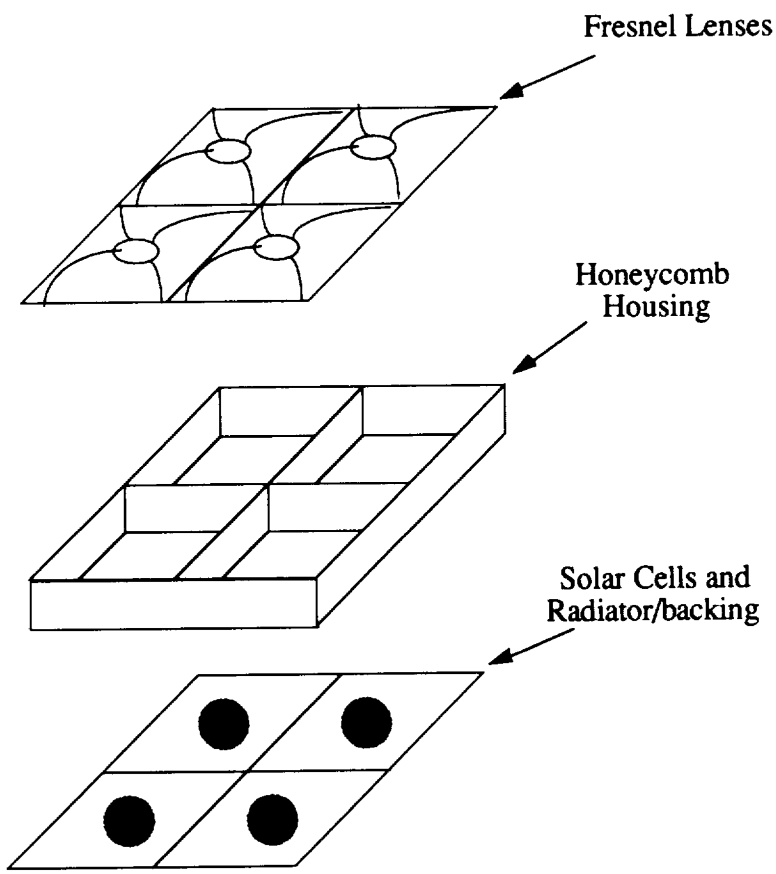


Figure 8-18
Lens Support Structure

The mass and thickness values for this array design are given in Table 8-13. The honeycomb structure and the lenses take up more than half of the weight allowance for this structure. This design does increase the expected lifetime of the actual solar cells because the Fresnel lenses act to protect the solar cells from solar proton flares and from micrometeorite impacts.

This array design requires that the assembled panels with the honeycomb structure and lenses be transported flat since the panels are fairly rigid and cannot be rolled or bent. The desired voltage output for each array is 32V. Using a factor of 1.1 for design considerations the design voltage is 35V. The array area dimension is 2m x 10m. This means that each array must hold 105 solar cells on $20m^2$ of backing. Five solar cells will be aligned along the width of the array with 21 solar cells along the length. This setup is shown in Figure 4. The thickness of each array panel will be slightly over 4 cm.

A second array design was investigated that did not use the protective Fresnel lenses. The solar cells will be close-packed on the array backing to make up for the sunlight concentration from the lenses. The solar cells, fully-assembled, occupy an area of approximately $4cm^2$. The area for one triplet leaving a 1cm gap between each cell, is then 2cm x 9cm or $18cm^2$. This would allow 6500 matched triplets per each solar array. This setup is shown in Figure 8-20. The triplets will be wired in series to produce a potential of 35V. There will be 185 of these sets of triplets wired in parallel.

Mass and thickness estimates for this design are given in Table 8-14 The mass of the array has been reduced dramatically with this design. Because this design does not employ any method of defending the solar cells from micrometeorite impacts or solar proton flares, the lifetime of each array will be greatly reduced. This design does allow for a much more

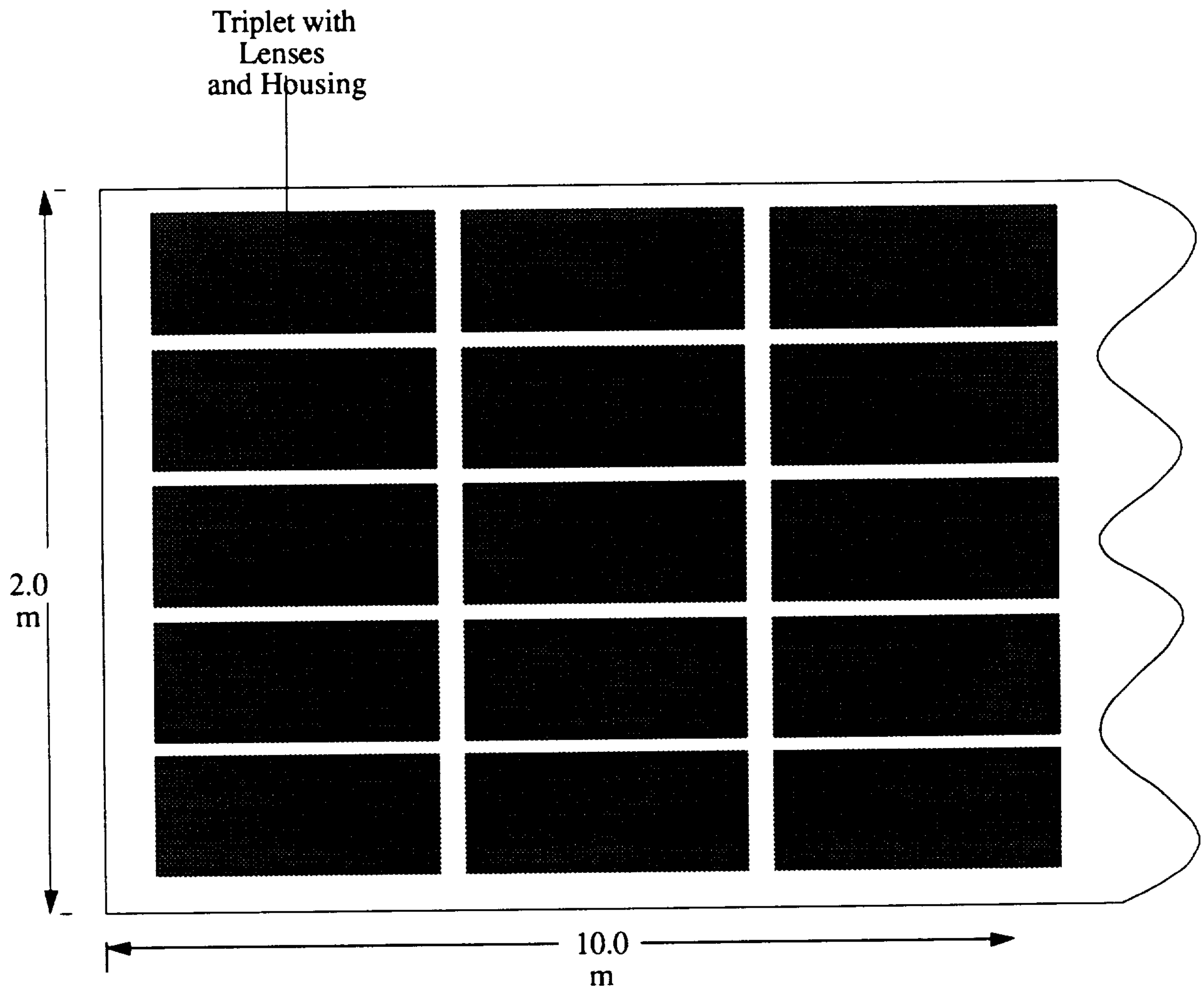


Figure 8-19
Setup for First Solar Array Design

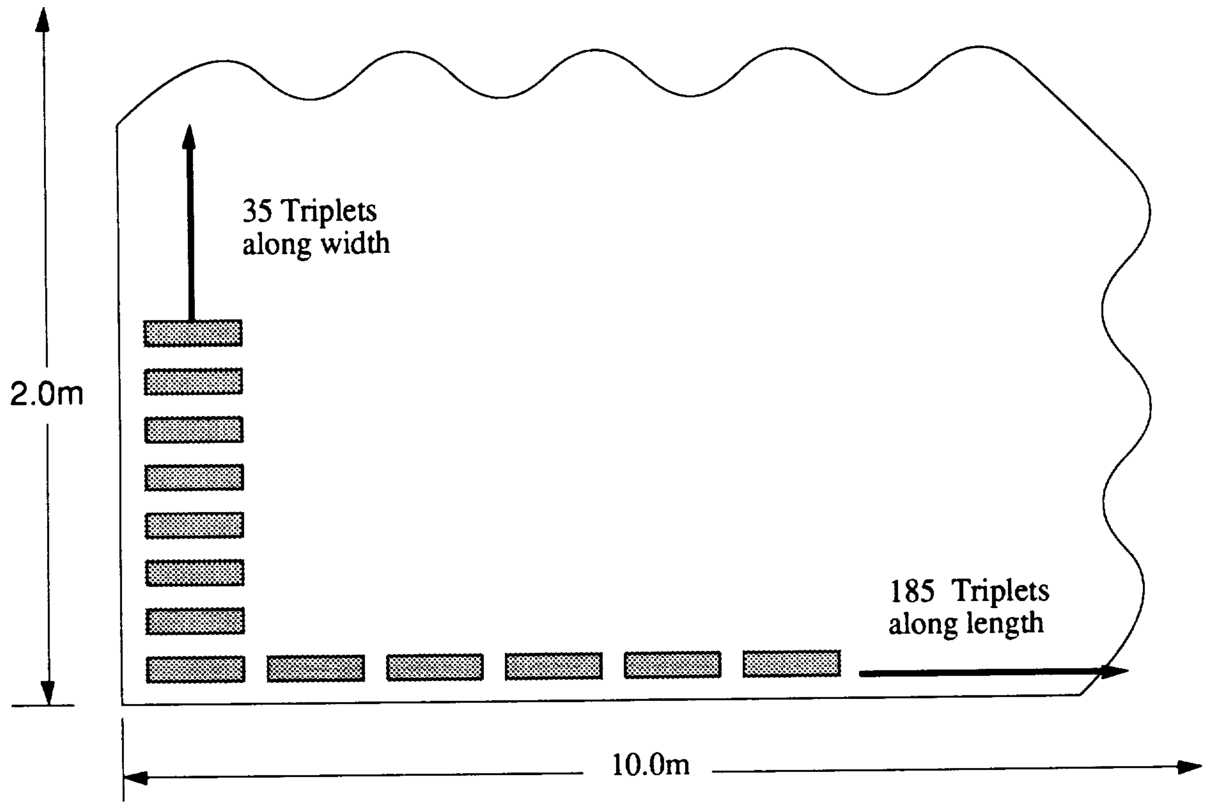


Figure 8-20
Setup for Second Solar Array Design

flexible panel that is much easier to assemble and replace than the previous design. The storage of these arrays is also a much simpler process. Since these panels are much thinner and more flexible, they can be rolled up for storage and transport.

Table 8-14 : Solar Array Estimates - Design 2

Array Component	Material	Thickness (mm)	Mass/Area (kg/m ^{2})	% Total Mass
Cell (assembled)	–	0.46	0.08	9.0
Radiator	aluminum	0.2	0.55	61.8
Radiator Paint	Polyethane	0.01	0.03	3.4
Miscellaneous	–	–	0.23	25.8
Total			0.89	

The second design that does not use the protective Fresnel lenses was chosen for application in the Solar Lunar Power Plant. Volume constraints for storage and shipment of the lunar surface became the limiting factors in the choice between the two designs. Other considerations were the relative simplicity for setup and replacement of the panels in the second design and the mass savings from the second design. In order to compensate for the increased reliability and lifetime of the first design, the number of solar cells per each array was increased dramatically. Anticipation of the need to replace panels from either design also favored the use of the second design for the arrays because of the ease in storage and replacemen

Array sizing and configuration

Total array area is determined by the power requirement, including compensation for all efficiencies and degradation factors, and by the efficiency of the solar cells. Minimum requirement set by the payloads is 19 kW for average daytime usable power and 27 kW nighttime power. For reasons mentioned earlier, our design parameters are set at 35 kW daytime usable pow$\eta_N$er and 35 kW nighttime power. Taking into account system losses, the total energy required from the array per one day is calculated by adding the energy required for 1 day's use plus energy for 1 night's use divided by the storage efficiency η_{night}:

$$E_{array} = E_{day} + E_{night} / \eta_{night} \tag{8-1}$$

Since energy equals power times time:

$$P_{array} (t_{day}) = P_{day} (t_{day}) + (t_{night}) P_{night} / \eta_{night} \tag{8-2}$$

and since the length of the daytime equals the length of the night: ($t_{day} = t_{night}$)

$$P_{array} = P_{day} + P_{night} / \eta_{night} \tag{8-3}$$

where P_{array} is the total power required from the array, P_{day} and P_{night} are the day and night power requirements of the user, and η_{night} is the fuel cell storage (electrolytic recharge) efficiency. Thus, the total power required from the array is 79.547 kW.

Given that the incident solar radiation is 1,360 W/m2, array efficiency is 23.5%, and Columbiad will have sun-tracking arrays (normal incidence at all times), the required array collection area can be determined by

$$A_C = P_{array} / (S\ (\eta_{array})) \qquad (8\text{-}4)$$

where A_C is the total collection area, η_{array} is the array efficiency, and S is the solar constant. This gives a total array area of 249 m^2. For redundancy, we will take an additional 50 m^2 of array, bringing the total to 300 m^2.

Panel Sizing and Set up

Individual panel dimensions are determined by structural limitations and ease of handling, primarily during set up and also during maintenance. For ideal power production, possibilitiy of panels shadowing each other should be minimized, preferably eliminiated. For best performance set up will be a 26 m long linear array. The array width is set to be 2 m. Each panel length will be 10 m for better structural weight and packaging. The panels are rotated about a central axis and mounted on legs, 1 m above the ground (Figure 8-21). This gives the total number of arrays to be 15 (13 on line and 2 in reserve) and the number of legs required to mount them is 30. The estimated mass of the panel support structure, including the legs, is 140 kg.

Tracking

Two schemes for panel tracking have been considered. One is to simply lay the panels flat on the ground, with no tracking. The advantage of this scheme is that this eliminates the need of motors and support structures and setiing up is trivial. However, the scheme requires twice the array area to obtain a given power output and since the power output is directly related to the sun angle, this would provide a unsteady power supply. Also, laying the panels on the surface would be associated with problems of lunar dust. Hence the untracked array set up is rejected.

Given the 5° tilt of the lunar axis, a single-axis system is sufficient to correctly position the panel. For better precision, the height of the mounting legs can be differentially varied. During the lunar day, the panels will be rotated 180° about the central axis along the lunar meridian. Considering an average 336 hour long lunar day, the sun only moves 0.54 °/hr, making a continous tracking unnecessary. For optimum performance, the panels should maintain direcitonal accuracy to within 0.5 °, thus a DC stepper motor is used to rotate the panels in 0.54 ° increments, one step every hour. These calculations may slightly vary depending on the landing site.

The motor are regulated by an open loop system; i.e. sun sensors are not included in the control loop. The motor, gears and controller will be encased in an insulated, tightly sealed housing to prevent from damage due to lunar dust. Each motor weighs roughly 5 kg, so for 12 motors the total weight is 60 kg.

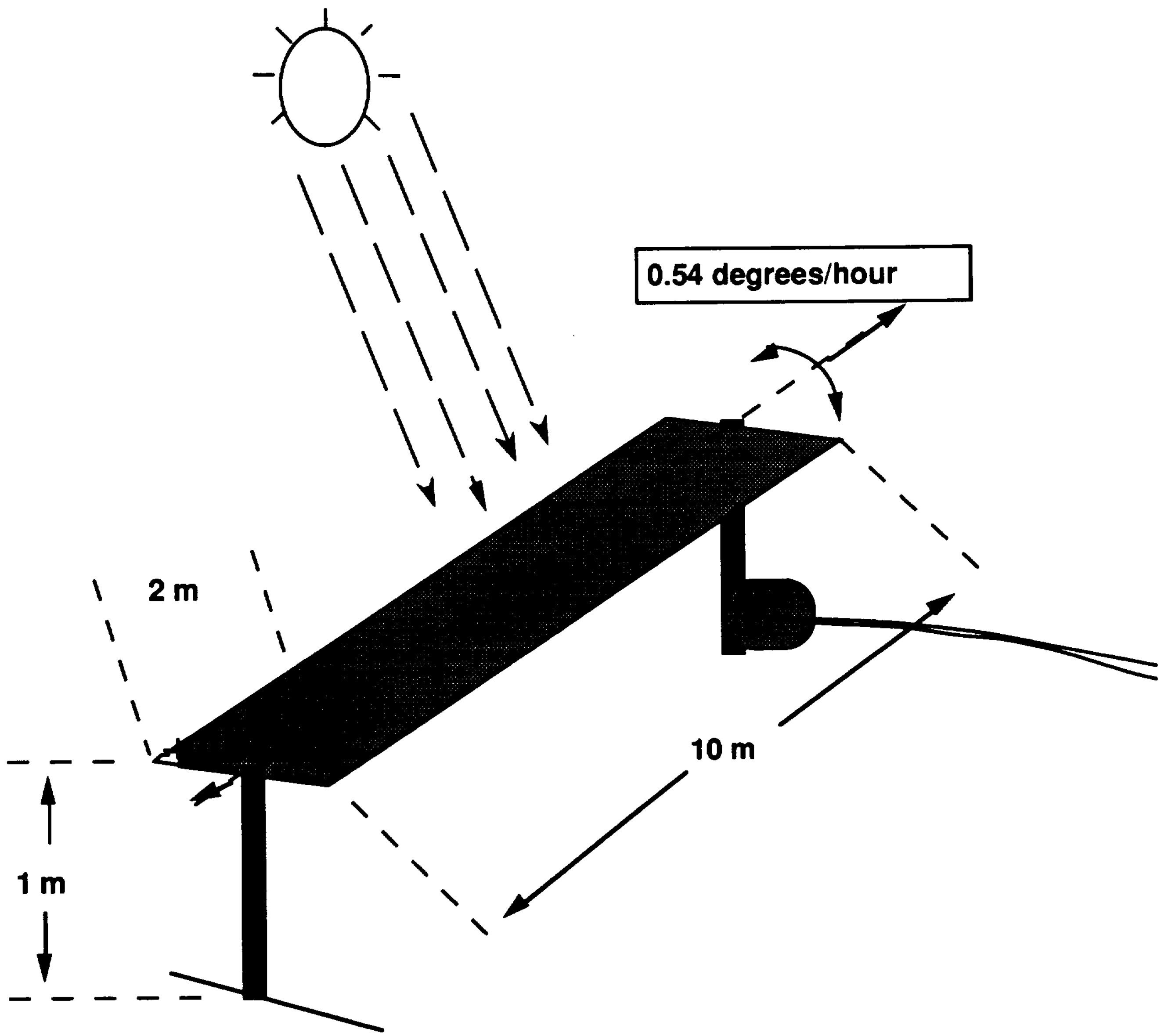


Figure 8-21
Panel dimentions and Set Up

8.2.2.3 Fuel Cell System Design and Description - Lunar Night Power Generation

8.2.2.3.1 Fuel Cell and Electrolysis Operation

Fuel cells generate electrical power by chemically combining two reactants, H2 and O2. A fuel cell operates in a fashion analogous to sealed battery cells except that a fuel cell's operation is constant and does not discharge as long as the reactants are continually resupplied and the products continually removed. H2 and O2 are fed into the fuel cell where they react, releasing electrical power and producing water which must be removed for continued operation.

The conversion of chemical energy to electrical energy proceeds in the following process: the hydrogen and oxygen flow through the cell, separated by an electrode layer. Oxygen ions pass through the electrode layer to the hydrogen side where a reaction occurs to produce an excess of electrons. These electrons are released to the anode of the cell, and so an electrical potential is established between the cathode and the anode. This potential is tapped by connecting an electrical load across the terminals, thus allowing current to flow.

The above process is performed during the night when sunlight is unavailable, to generate power for surface operations from the stored reactants. At the end of the night, most of the reactants will have been converted to water, and must be converted back to hydrogen and oxygen to provide power for the next lunar night. This is done in a process called electrolysis, which is basically the fuel cell reaction running in reverse. Instead of releasing energy from the cell, electrical energy is fed into the cell to dissociate the water back into its diatomic components. As previously mentioned, since electrolysis is done during the day, the solar array is sized to provide power for electrolysis (plus water vapor removal and liquefaction) in addition to the set requirement of 35000W continuous power.

8.2.2.3.2 Comparison of Two Fuel Cells

Two types of fuel cells have been developed with electrolysis capabilities or along with electrolysis units to create regenerative fuel cell systems. An alkaline fuel cell with an electrolysis subsystem has been developed by United Technologies Corporation, and a monolithic solid oxide fuel cell (MSOFC) with electrolysis capability has been developed by the Allied Signal AiResearch Division.

A comparison of the performance parameters of the two fuel cells was done to choose the type of fuel cell to be used in SLURPP. (See table 6-3)

Table 8-15: Fuel Cell Characteristics Comparison

	MSOFC	Alkaline
specific power (W/kg)	8	.147
volumetric power density (kw/l)	4	.118
operating temperature (K)	1273	330
discharge efficiency	60%	70%
electrolysis efficiency	66%	78.75%
stage of development	experimental	in use

The MSOFC cell is superior in terms of specific power and volumetric power density. However, the fuel cell hardware mass and volume is really quite small compared to the reactant mass and volume, so those parameters are actually not of relative importance. The cell which cuts down on reactant mass and necesssary solar array mass will actually be superior. Since the alkaline cell has a higher discharge efficiency, it requires less reactant mass to be processed for the same amount of energy output, so the alkaline saves substantial reactant mass. Furthermore, since the electrolysis efficiency is higher, it requires that less power during the day be diverted to the electrolysis units, so the arrays can be smaller in conjunction with the alkaline fuel cell and still meet the day power requirement. Next, the operating temperature of the alkaline cell is much lower than that of the MSOFC cell, so the thermal control problem of the fuel cells should be lessened in using the alkaline cell. Last, the alkaline fuel cell is currently used on board the space shuttle orbiter and has never experienced a serious in-flight failure, while the MSOFC is still much closer to the experimental stage. For all the above reasons, it was deemed sensible to include the alkaline fuel cell in the design of SLURPP (and in all other power applications of Project Columbiad which call for fuel cells.)

8.2.2.3.3 Fuel Cell Apparatus Design

The alkaline fuel cell system consists of a stack of layered cells, each layer holding a potential difference from the top surface to the bottom surface by virtue of the internal chemical reaction. The typical voltage for one of these layers is approximately one volt, so it is possible to build up to design voltages by stacking the layers in series. For extended life of the cell, it can be run at a low-level current density of .323 A/sq.cm as compared to a maximum of 2.15 A/sq.cm. This extended life is desirable for SLURPP, so the low current design is chosen. This design gives a volumetric power density of 118000 W/cu.m and a specific power of 147 W/kg. For a base area of 29cm X 29cm, a stack of cells would carry 273 A. To build up to the surface hardware common bus voltage of 32Vdc, one would stack 32 cells in series, so the power output from a single stack would be 8750W. Four such 32V stacks in parallel will give the night power continuous output of 35000W at a voltage of 32 volts dc. Another such stack with slightly more base area will supply the backup power of 10000W as requested by Surface Payloads. The total SLURPP fuel cell system will occupy a volume of 0.381 cu.m (29cm X 145cm X 88cm) and have a dry mass of 306 kg.

8.2.2.3.4 Cryogenic Reactant Storage/ Liquefaction

By supercooling a reactant gas down to the liquid phase, and maintaining refrigeration of the fluid, one can cut down tremendously the mass of the storage tanks as opposed to the massive tanks which are required to hold the gaseous phase at high pressure, as was demonstrated in the University of Washington Solar Plant report. It has been determined that the rocket propellants in all of the propulsion stages of Project Columbiad will be stored cryogenically. To cut down on the number of cryogenic systems which must be designed, it was decided that the reactant tanks of SLURPP will be integrated with the propellant tanks of the PLM, the stage which carries SLURPP to the lunar surface. Since both the RL-10 engines and the fuel cells use LOX-LH2, it was a clear choice to combine the tanks into one pair of tanks. This decision also makes use of the PLM structure and tanks, which would otherwise be a useless monument after the Precursor landed on the moon.

The only difficulty of integrating the propellant tanks with the fuel cell reactant tanks in the PLM is that the RL-10 engines would like their LOX and LH2 fed to them at about 340000 Pa, while the fuel cells require a minimum input pressure of 689285 Pa, a much higher pressure. To make up for this difference, a technique called static heating is implemented to raise the reactant pressure up from the engine's desired propellant pressure to the fuel cell's desired reactant pressure. Thus the PLM tanks stay at their low pressure, lower mass design.

Static heating is a technique used on board the space shuttle. This process takes a cryo fluid flow and raises the flow pressure in the following way: A portion of the flow is diverted through a heater and vaporized to higher pressure, and then returned to the main flow. A high-pressure boundary layer of vapor forms on the outsides of the flow and drives the internal liquid pressure up. This scheme safely leaves most of the cryo fluid in the desired liquid phase, and allows the fluid pressure to be adjusted as a function of the amount of heat injected into the flow. A typical static heater uses about 1000W of power.

The cryogenic system of the PLM-SLURPP may differ from that of the other stages in that while the other stages are simply passive tanks, the PLM-SLURPP tanks may be fitted with active refrigeration so that while the tanks are used and reused as a part of SLURPP on the lunar surface, they will be able to refrigerate and store their reactants for the extended time of the lunar day. The more feasible and likely option is to design the cryogenic tanks passively like the other stages' tanks, which means that one will expect a .175% boiloff of the original mass of reactants per month. Since the reactants are cryogenically stored in the tanks only half of the time, this translates to a loss of 10.5% of the reactants after ten years of power plant operation. After ten years, the night continuous power level would be reduced by 10.5% to about 31325W, or else a vehicle could be sent from the earth to replenish the lost reactants. This will probably be deemed to be an acceptable loss rate in light of the passive system simplicity it affords.

Two final concerns for the fuel cell reactant storage system are those of reactant drying and liquefaction. During the day, as previously mentioned, the electrolysis units will be "recharging" the night power storage system by dissociating the water down to diatomic oxygen and hydrogen. As these gases leave the electrolysis units, they may still contain a small amount of water vapor which was not dissociated. As they are on their way to the liquefaction unit and ultimately to the cryo-storage tanks, these gases must be dried out or else the water vapor will freeze and lead to blockage problems. A drier unit must be included in the design to remove the vapor. A typical such drier which can handle a 35 kW unit will use .3 kW continuous during the entire day and weighs in at 28 kg.

After the gases are dried, they must be liquefied before placement in the cryogenic storage tanks. A typical H2 liquefaction unit which can handle a 35 kW unit requires 3.88 kW continuous day power, and has a mass of 428 kg. A typical O2 unit requires 1.84 kW continuous day power and weighs in at 136 kg.

8.2.2.3.5 Water Storage

During night operation, the fuel cells will be producing water which must be stored for electrolytic reprocessing. This water will be held in an expandable bladder tank on board the PLM-SLURPP module and will be warmed during the lunar night with heat from the discharging fuel cells, hence the water is not expected to freeze.

8.2.2.4 SLURRP Thermal Control Considerations

Energy that is absorbed by the solar cells that is not converted into electrical energy is dissipated as heat. It is desirable to maintain the solar cells at low temperatures to keep them operating at higher efficiencies. It is therefore, necessary to dispose of the excess heat that is generated. When investigating a thermal control system for the solar panels, the physical characteristics and the location of the solar panels must be taken into consideration. The panels occupy a very large area. This means there is a very large surface area available for cooling. The panels are also located on the surface of the Moon which means that the surface area could be used to radiate the heat directly into deep space.

Thermal control issues for the fuel cells are similar to those of the solar panels, but the physical characteristics and location of the cells is quite different for the fuel cells. The fuel cells are very compact. There is a limited amount of surface area available to conduct heat away from the fuel cells. All of the fuel cells will be stored within the walls of the spacecraft at all times. This means that it will be necessary to transport the heat away from the fuel cells and out of the spacecraft. The fuel cells are also more sensitive to any fluctuation in temperature so this system must be more accurate.

The overall concerns of the design of the thermal control systems was to try and keep them simple and reliable. Passive systems were preferred to more complex ones that would require excessive instrumentation or materials.

8.2.2.4.1 Radiator Design for Solar Cells

The solar array panels provide enough surface area that the actual backing of the solar cells could be used as a radiator. This would effectively utilize the area of the arrays. The radiator would act as the backing support for the solar cells, and it would still remain relatively thin. This design is very simple. The cooling system is passive and there is no need to worry about transferring the heat away from the cells to an external radiator so there is no need to investigate any coolants.

Material needed to chosen for the radiator backing. In order to select the appropriate material for the radiator, the required material characteristics were first compiled. The material needed to be a good conductor of heat to insure that the cooling is rapid and even along the entire expanse of the panel so the material does not warp. Another consideration is that the material have a relatively low density and a reasonably low cost. Due to the large fluctuation of the temperature on the surface of the moon, it is necessary that the material have a high heat capacity and high melting point. The material should be malleable so that it can be rolled without breaking or cracking and so that the thin sheets can be manufactured without much effort. The radiator material should not react naturally with any of the material in the lunar soil.

Three commonly used conductor materials were considered for use in the radiator backing. They were aluminum, copper and silver. Their relevant properties are listed in Table 8-16 Silver and copper both have much higher thermal conductivities compared to aluminum, but they also have much higher densities. The lowest melting point temperature is still much higher than the daytime temperature of the moon. All of the materials can be manufactured with a reasonable ductility. The coefficients of thermal expansion for the materials shown are all very close and none of the materials presents a distinct advantage over the others. The cost of the silver is one hundred times that of either the copper or the aluminum.

Table 8-16 : Material Properties

Material	Thermal Conductivity (W/cm•K)	Density (kg/m ^{3})	Ductility	Melting Point (K)	Thermal Expansion (MK^{-1})	Cost ($/tonne)
Aluminum	2.36	2,650	0.1-0.5	933	24	1180
Copper	4.01	8,960	0.5-0.9	1356	17	1330
Silver	4.28	10,500	0.6	1235	19	130,000

Aluminum was chosen for the radiator backing because it presented a better conuctivity to mass ratio than the copper or aluminum and because using copper or silver would only slightly increase the performance capability of the radiator. The equilibrium temperature of the aluminum radiator can be found using equation 8-5 The total radiated energy in this case would be 76.5% of the incoming solar radiation (1358W/m^2). The emissivity of uncoated aluminum is 0.05. This yields an operating temperature of 778K. This is too high. In order to increase the emissivity of the back of the radiator panel, the aluminum will be coated with a thin layer of black paint. This will increase the emissivity to 0.874.

$$T_e = \left[\frac{\alpha Q_s}{\varepsilon\sigma}\right]^{\frac{1}{4}} \tag{8-5}$$

With this emissivity the equilibrium temperature is reduced to 370K. The necessary thickness of the aluminum radiator is found using equation 8-6 Since the equilibrium temperature is so low, the required aluminum thickness is minimal. The thickness chosen for the radiator was 0.2mm.

$$q = \frac{k}{\Delta x}(T_e - T_2) \tag{8-6}$$

The resultant radiator mass for one array is 11 kg. The breakdown and total mass for one 2m x 10m array is given in Table 8-17. The total mass for one array does not include the support structure mass. This is the total mass for the actual solar array sheet.

Table 8-17 : Array Mass Breakdown

Array Component	Mass (kg)
Cell (assembled)	1.5
Radiator	11
Radiator Paint	0.5
Miscellaneous	4.5
Total	17.5

8.2.2.4.2 Fuel Cell Thermal Control

The fuel cells are contained within the walls of the spacecraft. There is a three step process involved with the removal of excess heat from the fuel cells. A path should be provided o conduct the heat away from the fuel cells to the outer wall of their encasement. This excess heat needs to be removed from the outer wall of the encasement to the outer wall of the spacecraft. A radiator needs to be setup to take the excess heat from the outer wall of the spacecraft and radiate it out into deep space.

To remove the excess heat from the actual fuel cells, an aluminum casing can be made to contain the fuel cells and conduct the excess heat from discharging away from the cells. In order to do this, an aluminum casing can be manufactured to fit the fuel cells such that each fuel cell base will be in contact with the flat base of the casing. The fuel cells operate at 330K with a power output of 20kW and with a 30% loss. The power that is lost as heat is 9kW. The thickness of the base plate of the aluminum casing is 2.0mm.

To cool down the outer wall of the base plate, a coolant can run past the outer wall. For simplicity, water was chosen as the coolant. It has a relatively high specific heat and problems with viscosity and flow will be minimized. The needed mass rate for the water can be found using equation 8-7. The mass flow rate for the water is 72g/s, or 72cm^3 /s. The cooling of the base plate will require a very thin film of water to be run across the bottom side of the casing. The water will need to be pumped through a cycle that runs it across the bottom of the base plate for the fuel cells up to the outer wall of the spacecraft, through a radiator, back through the wall of the spacecraft to the fuel cell casing. This would require 5m of pipe for the transfer of the water. In order to insure that the heat being removed from the cells is deposited outside of the spacecraft, the pipes also need to be insulated. The pipes can be coated lightly with a thermoplastic for insulation.

$$Q = \dot{m}\overline{C}\Delta T \tag{8-7}$$

The external radiator can be an extension of the pipes to the outside of the spacecraft. The exposed length of pipes should be coated with black paint to give it a higher emissivity. Mass estimates for this cooling system are given in Table 8-18.

Table 8-18 : Cooling System Mass Estimates

Component	Mass (kg)
Casing	5
Pipe	2.3
Water	0.1
Water Pump	10
Total	17.4

8.2.2.5 Mini-SLURPP Design

As a special consideration for the time profile of the Project Columbiad Mission, a small autonomous solar power plant needed to be designed to supply the surface hardware with power during the interim between the Precursor vehicle landing and the Piloted vehicle landing. This time period could last up to one year, and power would not otherwise be available because the SLURPP solar arrays must be set up by astronauts. Power needs include sensors, communications, and interior thermal control, and these needs must be minimized during this "hibernation" period of waiting for the astronauts' arrival.

As one option for supplying power, the SLURPP fuel cell system could be slowly discharged during the year, offerring approximately 3 kW of power continuously. However, when the astronauts arrived just before nightfall(as is planned), the fuel cell system would be completely discharged, and could not be recharged until the following lunar day. Even slightly less than full night power is unacceptable because the first night will be the time of the largest construction effort, of burying the BioCan with regolith and other power-consuming activities. To preserve the night power capability, another source of energy needed to be found for supplying the hibernation period.

Another source of energy during the hibernation period is the sun, so the provision has been made for a pair of solar panels to autonomously unfold, in "accordian" style, from the sides of the power section of the PLM, and partially track the sun using a simplified articulated pair of hinge joints. Each of these panels is 2m X 10 m, and together they output an average of 5.28 kW of power. Of this, 2.5kW must go to electrolytic recharge of a portion of the SLURPP fuel cells for night power, so the net continuous usable power output is approximately 2.5 kW continuous for this system, a power supply which can easily handle sensor and communication needs as well as thermally maintain the environment of the BioCan.

8.2.3 SLURPP Complete System Description and Mass/Volume Breakdown

Table 6-4 lists all the salient parameters of the Solar Lunar Power Plant, and gives an overview of all the hardware components.

Table 8-19 : SLURPP Salient Parameters

Day power: 79547W
Day Usable Power: 35000W
Night Power: 35000W

Solar Cell Effic.: 23.5%
Total Solar Array Area: 249 sq. m
Mass/Area of Array: 1.1 kg/sq.m
Total Array Mass: 273.8 kg

Alkaline fuel cell discharge efficiency: 70%
electrolysis efficiency: 78.57%
specific power: 147 W/kg
volumetric power density: 118000 W/ cu. m

Night Storage: reactant mass: 5030.7 kg
fuel cell hardware mass: 238 kg
Water vapor removal unit: 28 kg
Liquefaction units: H2: 428 kg; O2: 136 kg

Backup Power: 10000W for 366 hours
backup fuel cell mass: 68kg
backup reactant mass: 1438kg

Total SLURPP Regenerative Fuel Cell System Mass: 6774 kg
Total Fuel Cell Hardware Volume: 0.381 cu. m
Total Fuel Cell System Volume (reactants + cells, night storage + backup): 15.54 cu. m
System Mass Subtotal (Array, Night Storage F.C. System, Backup Power): **7048 kg**

Cryogenic Storage System: (Night Storage plus Backup Power)
mass of O2: 5749 kg; cryo density O2: 1140 kg/cu.m; vol. of O2: 5.04 cu. m
mass of H2: 718.7 kg cryo density H2: 71 kg/ cu. m; vol. of H2: 10.12 cu. m
Required LH2 Purity for fuel cells: 0.99990; Required LOX Purity: 0.99989
H2O volume when totally discharged: 6.47 cu. m

8.3 Rover

8.3.1 Requirements

The Lunar Rover is required for surface transportation of personnel and payload on the Moon. The required range of the rover is 100 km. The maximum limit on the distance from the habitat is set by the walkback capability of the astronauts. The vehicle has to carry two personnel nominal mission and with provision for carrying the other two astronauts during emergency. Including the astronauts, the rover should be able to carry a minimum of 500 kg of payload. Various other surface operations equipment is dependent on the rover for their mobility and operation, hence proper attachment mechanisms have to designed. The vehicle should be able to be remotely driven, at least within line of sight. Since there is no redundant vehicle, the rover system, - particularly its drive mechanism, has to highly reliable.

8.3.2 Design

Due to superior performance shown by the Apollo LRV, the Rover is designed using the LRV as a baseline. It is a six-wheel driven, four-wheel steered vehicle. The entire base divided into four sections. The last section being a four-wheeled, detachable Wagon trailer. The fully deployed rover is 5.5 m long and 2.5 m wide. The height of the vehicle is 2 m, excluding its antenna. The vehicle is powered for 150 kms, nominal mission range being 120 km at an maximum velocity of 20 km/hour. To ensure the walkback capability of the astronauts, all missions are limited within a 50 km radius of the habitat. The maximum mission duration is 8 hours. The vehicle is unpresserized, but the astronauts can hook up their EVA suits to the PLSS packs on-board the rover. The astronauts' PLSS backpacks are held in reserve for off-the-vehicle activities and for emergency procedures.

8.3.2.1 Structural Design

There are two primary structural concerns with the Rover. The first, is the volume constraints of the vehicle. The rover must be contained with the payload bay of the PLM module. To allow the vehicle to be of acceptable size, it is necessary to fold the vehicle in a collapsed state. Once on the Moon surface, the vehicle will be expanded to its final a rigid state. The vehicle is designed to only unfold once. There is no requirement for multiple breakdowns and assemblies.

The second concern is weight. Since payload capacity is severely limited, and with the requirement to carry additional crew provision and stores, mass optimization is critical. The design presented below is of only preliminary efforts. The amount of iteration and detailed design that is desired for such a critical component of the mission has not been performed due to the time constraints of the preliminary design phase of the project. The design of the lunar vehicle is an area that merits considerable attention during critical design.

The following constants are used in the design of the lunar Rover. See Table 8-20

Table 8–20 : Constants for Lunar Rover Vehicle Mass Calculation

Constants	
Pi	3.1416
Composite Density	1490 kg/m^3
Aluminum Density	2700 kg/m^3
Steel Density	7600 kg/m^3
Semi-monocoque Coefficient	60%

The lunar Rover is broken into four major components. The base plate, which forms the structural heart of the vehicle since all other components are mated to it. The base plate was design using the semi-monocoque coefficient described in Vol. II 2.1. The wheels, are obviously hollow, and their design has been adopted from the Apollo mission. They are steel wire meshes with circular springs built into the rim. The third component, is the folding mechanism, which consists of the devices that allow the vehicle to be furled up, and then deploy once on the moon. This is assumed to be a percentage of the base plate weight. The final and fourth component is the drive train mechanism. This consists of the axle mechanisms and the gear boxes that allow the electric motors to drive the wheels. In addition, this includes the steering mechanism. It is assumed to function as a function of the total wheel weight.

The design procedure is very straightforward, and most of the values in Table 8–21 are self explanatory. The total weight of the structural mass of the lunar vehicle is 291kg.

The wagon, which is used as a utility hauling vehicle, and also to move regolith from the Collector to the Conveyer, is designed according to the parameters shown in Table 8–22. Their is a trailer-type hitch between the lunar vehicle and the wagon. The actual joint is shown in Figure 8–22. This joint has full three degree-of-freedom motion. This should ease and travel over the rough lunar terrain.

Table 8-21 : Lunar Rover Design Parameters

Lunar Excursion Vehicle	
Width of Vehicle	2 m
Length of Vehicle	4 m
Base Plate Equivalent Thickness	0.01 m
Base Plate Mass	129.6 kg
Folding Mechanism Ratio	15%
Folding Mechanism Mass	19.44 kg
Number of LEV Wheels	4
LEV Wheel Diameter	0.6 m
LEV Wheel Width	0.25 m
LEV Wheel Equivalent Thickness	0.005 m
LEV Wheel Solidity Ratio	50%
LEV Wheel Mass	20 kg
Total LEV Wheel Mass	79 kg
LEV Drive Train Mechanism Ratio	80%
LEV Drive Train Mechanism Mass	63 kg
Lunar Rover Mass	291 kg

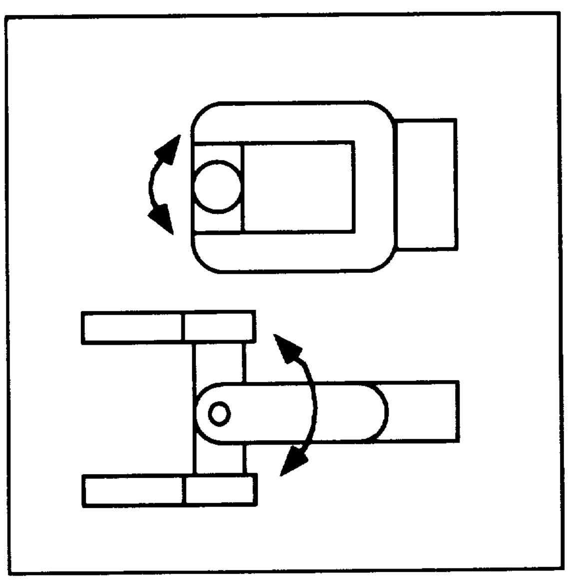


Figure 8-22
Lunar Rover Joint

The Wagon has nearly the same components as the lunar rover. There is however, one difference. It requires a dumping mechanism on the bottom of the vehicle that will allow the regolith dirt to be dumped on to a conveyor belt to be deposited on the habitat. This is assumed to be a percentage of the total wall mass.

Table 8-22 : Wagon Vehicle Design Parameters

Wagon Design		
Wagon Volume	1	m^3
Wagon Length	1	m
Wagon Width	2	m
Wagon Height	0.5	m
Wagon Container Area	5	m^2
Equivalent Wall Thickness	0.01	m
Container Mass	81	kg
Dumping Mechanism Ratio	10%	
Dumping Mechanism Weight	8.1	kg
Number of Wagon Wheels	4	
Wagon Wheel Diameter	0.6	m
Wagon Wheel Width	0.25	m
Wagon Wheel Equivalent Thickness	0.005	m
Wagon Wheel Solidity Ratio	50%	
Wagon Wheel Mass	20	kg
Total Wagon Wheel Mass	79	kg
Wagon Drive Train Mechanism Ratio	80%	
Wagon Drive Train Mechanism Mass	63	kg
Wagon Mass	231	kg

The total mass of the Wagon structure is 231kg.

8.3.2.2 Layout

Figures 8-23 and 8-24 show the top and side schematic view of the Rover. The vehicle is divided into four sections. The last section is the four-wheeled Wagon. This section is detachable from the rest of the vehicle. The front section carries the most of the communication, navigation equipment, the camera, video recording equipment, the headlights, some batteries and as with all other driven wheels, the front wheels have their drive mechanisms next to them. The next is the crew section. This portion of the vehicle includes two chairs in tandem with an instrument panel in front of them. The Rover can be driven from sitting in either of the chairs. A joystick, on the common armpad between the chairs is used for steering the vehicle. The on-board PLSS packs, along with the equipment for recharging the PLSS packs on the astronauts' suits are located under each chair. The third section carries the batteries, some of the drive mechanism, tools and accessories and the antennas. In addition to that, a part of this section can be used as storage space for payload. This space will be used particularly in long trips when pulling the Wagon can reduce the efficiency of the Rover.

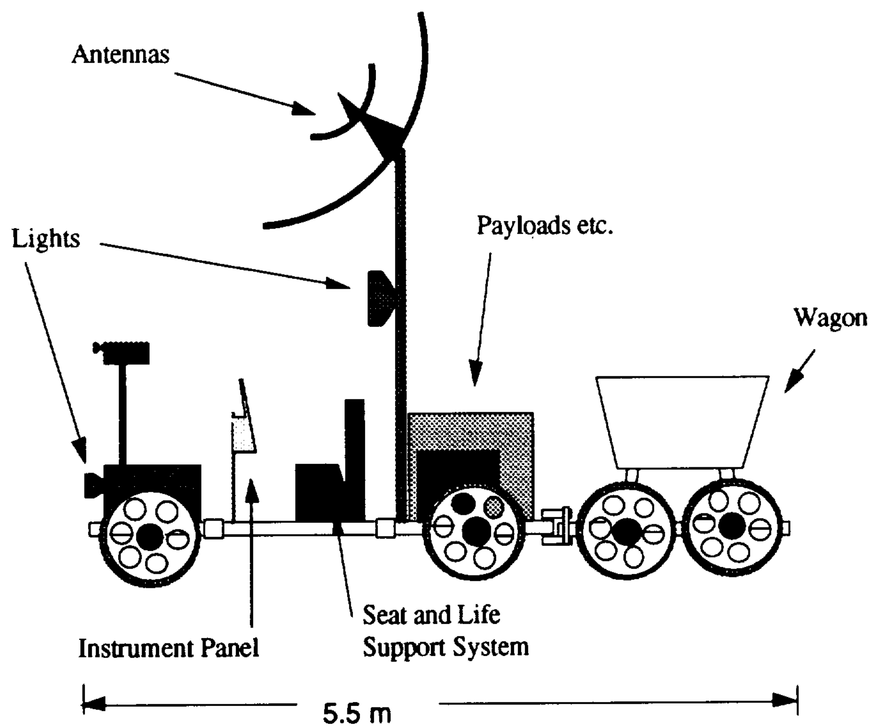


Figure 8-23
Lunar Rover - side view

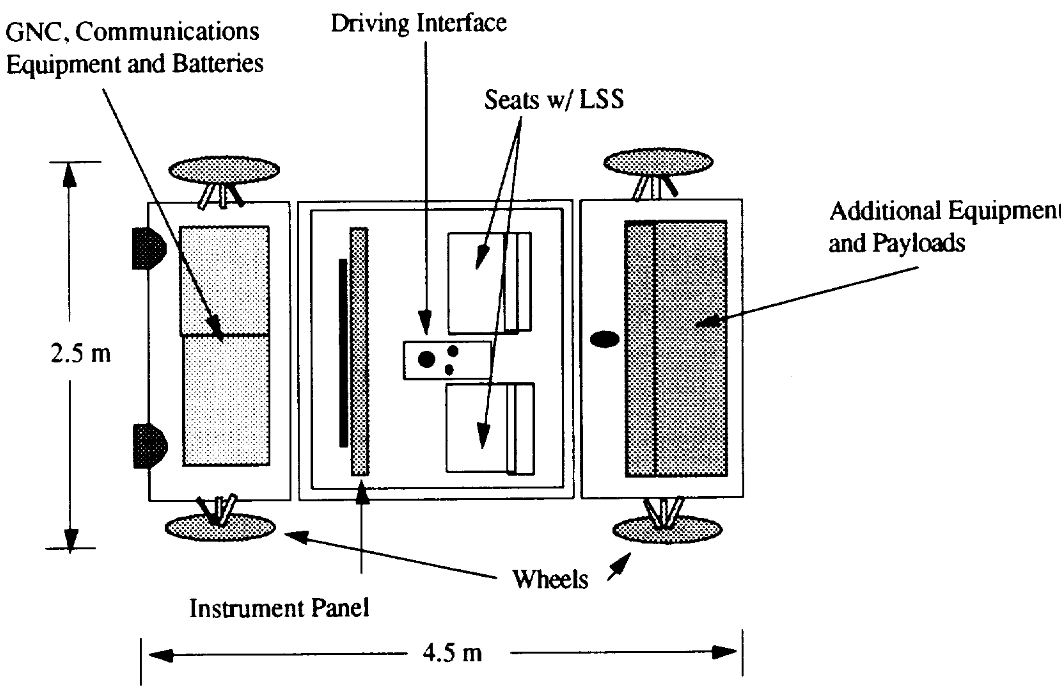


Figure 8-24
Lunar Rover - top view (without the Wagon)

Figure 8-25 shows the stowed configuration of the Rover including the Wagon. Since storage space in the cargo bay of the PLM is limited, the stowed volume of the Rover is kept around 10 cubic meters. Folded once from the joint between the second and the third sections, the entire Rover can be packed into a 3.25 m x 2 m base, with a height of 1.6 meters. The wheels are folded inwards, above and below the Rover. The stowed Rover will be provided with four strolling wheels. Upon reaching the Moon, during deployment it is rolled out through the cargo bay hatch, along a ramp, on these wheels. The top fold flips out and deploys as the first two sections of the Rover. After the front wheels fold out and lock in position, they touch the surface and support the Rover. The rest of the wheels along with the chairs, instrument panels and Wagon legs etc. deploy after that. Next, the antenna boom is stretched out and the antenna dishes are folded out to complete the Rover deployment.

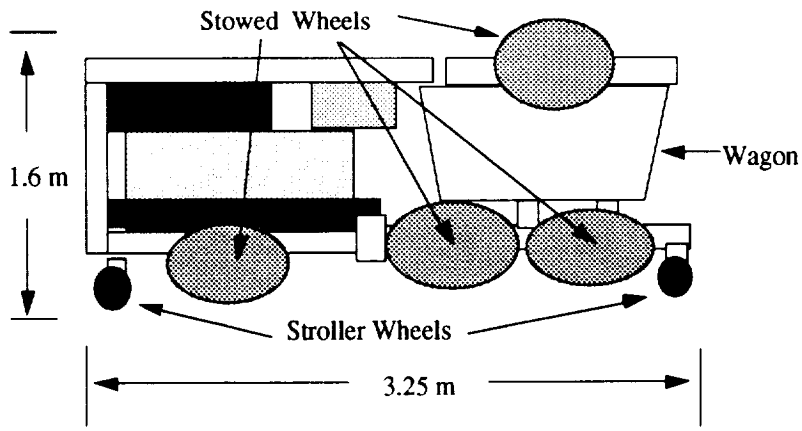


Figure 8-25
Lunar Rover - stowed configuration

8.3.2.3 Equipment

8.3.2.3.1 Power and Thermal Control

The power requirements of the rover is distributed as follows :

Drive Motors	5 kW
Instrumentation	1 kW
C3	0.5 kW
Crew Systems	0.5 kW
GNC	0.5 kW
TOTAL	7.5 kW

The lunar rover has a unique set of power system requirements. Similar to the requirements of Apollo's LRV power units, the Columbiad rover power plant should output high power and store large amounts of energy for lowest possible weight, and should be of the sealed cell type to avoid the complexities of cryo, plumbing, etc. of fuel cell systems. Moreover, unlike Apollo's LRV, the Columbiad rover batteries must also be rechargeable so the rover is reusable for an extended life. The Na-S sealed cell, still in its developmental stage, meets all the above requirements with rechargeability, at the highest specific energy (210 Whrs/kg) available for any sealed system.

The Columbiad rover calls for a power capability of 7.5 kW for 8 hrs, or a total energy of 60 kWh. With a safety factor of 1.1, the total on-board capacity is 66 kWh.The sodium sulfur battery packs have a total mass of 316 kg and a total volume of approximately .058 m^3. The total stored energy is distrbuted in smaller banks and placed in different locations on the vehicle. Within the banks, smaller units are connected in parallel such that in case a failure, one small pack can handle the peak load.

Thermal control concerns on the rover will be accomodated passively through the placement of radiative heat sink fins on thermal hot spots, such as the drive motors and battery packs.

8.3.2.3.2 Crew Systems

For the sake of modularity, the PLSS units on the rover will be comprised of virtually identical components to those on the primary lunar EVA PLSS. These will supply 8 hours of oxygen at 0.34 atm. Unlike the Ames PLSS, the rover units will not include the 1/2 hour emergency oxygen, as the primary supply serves doubly as en-route- and emergency breakdown-oxygen. Astronauts will simply engage an umbilical from the rover system to his suit PLSS, which then automatically disengages the suit PLSS. Like the suit PLSS, thermal circulation, water supply, and battery power is also supplied, bypassing the suit unit when attached via umbilical. Another feature which varies from the EVA suit PLSS is the detachability of the oxygen and water supply tanks which can be filled inside the lunar habitat using the PLSS recharge system and a tank adapter.

The final rover PLSS feature which varies significantly from the EVA suit PLSS is the ability for astronauts to transfer oxygen from the rover tanks to the EVA suit tanks. In the event of irreparable damage or malfunctions which would take longer than oxygen supplies would last, the astronauts would be forced to walk back to the habitat. The extra oxygen present in the rover tanks could become essential if the breakdown occurred far from the habitat. Therefore, along with the rover PLSS there exists a compressor system which enables an astronaut to transfer oxygen to his own PLSS from the rover.

8.3.2.3.3 Communications

The lunar rover will have a line of sight communication link with the habitat and a direct link with the Earth. Details of the communications system found on the lunar rover are discussed in Volume II sections 4.2.6 and 4.3.4.

In case of an emergency walk-back from the rover malfunction site to the habitat, there will be no communication link between the astronauts and the Earth. The astronauts will be able to communicate with each other through their spacesuits and once within the line-of-sight range of the habitat they will be able to reestablish their communication link with the habitat.

8.3.2.3.4 Navigation

Since a significant portion of the rover's opreation will take place beyond the line-of-sight of the BioCan, the rover needs to have on-board navigation equipment. The easiest form of navigation possible is inertial navigation. Using the rover's computer and a INS, it is possible to integraate INS output in real-time to establish its position. This information can then be displayed on a screen in the instrument panel. In order to zero-out the drift rate error of the INS gyros, the rover has to stop at certain intervals and take zero-velocity readings.

Within 1 km range of the habitat, the rover will switch to a active transponder beacon guidance. The beacon signal allows the rover to determine its bearing and range precisely.

In the unlikely event of a major rover malfunction, the astronauts will have to walk back to the habitat. For this reason, the computer and the INS package can be unplugged from the rover and carried by the astronauts. Including a battery pack, the portable system has a mass of only 25 kg.

8.3.2.3.5 Tools and Accessories

All additional equipment is divided into two catagories : (i) General Scientific Equipment and (ii) Basic Repair Tools and Spare parts

Table 8-23: Tools and Accessories for the Rover

General Scientific Equipment		
Surface Sample Collection Equipment	Tongs, Hammers, Scoop, Rake, Drill, Core Tubes, Sample Bags etc	95 kg

Recording Equipment	Film Cameras, Video Cameras, Films, Video Cassattes, Portable Spotlights etc.	50 kg
TOTAL	(Volume 0.65 cu.meters)	145 kg

Basic Repair Tools and Spare Parts		
Spare Parts	Wheel (2), Lights, Fenders, Joints and attachmentsi etc.	100 kg
Repair Tools	Wrenchs, Hydralic jacks, screwdrivers etc.	50 kg
TOTAL	(Volume 1.1 cu. meters)	150 kg

Total mass of the tools and accessories is 285 kg and it takes up roughly 1.75 cubic meters of volume. All these equipment is stowed behind the astronauts' seats, in the third section of the rover.

8.3.2.4 Overall Specifications

Performance Characteristics	Maximum Range	150 km
	Nominal Range	120 km
	Maximum Radius	50 km
	Maimum Mission Duration	8 hours
	Maximum Speed	20 km/hour

Mass Estimates		
	Structure	522 kg
	Diving Motors	80 kg
	Communication equipment	98 kg
	Batteries	316 kg
	Crew Systems	65 kg
	Additional Equipment	295 kg
	TOTAL	**1376 kg**

8.3.3 Operations

8.3.3.1 Issues in Driving

The Apollo data on lunar surface has concluded that the lunar soil has a constant bearing strength. Thus it can be assumed that the Rover's mobility will not be hindered by the presence of unusually soft soil. The principal barriers that are expected are steep slopes and boulder fields at the rims of fresh craters, portions of the walls of rills, and parts of fault scraps. To minimize long circuitous maneuvers to avoid obstacles, it is assumed that all traverses whether for science, resource exploration, or base logistical support will be preplanned to some extent. Initially, traverses will have to be planned and practiced with the thoroughness of Apollo mission traverses, including a lot of ground support. Once the operating characteristics of the vehicle are well known, planning can be limited to a detailed traverse route and an overall timeline. The maximum driving speed will probably be more of a function of the terrain than the performance factors of the vehicle. Due to low lunar gravity, the Rover is likely to become airbourne for a significant time during a rough traverse.

Illumination of the surface is also a major issue. The Rover will need a minimum illumination of the surface ahead of it to allow time for detection and avoidance maneuvers or stopping before running into various obstacles. On-board sensors should automatically turn on artificial lights whenever the surrounding natural light level goes below a certain limit. Obviously for nighttime traverses, the artificial lights become more crucial. Driving through high angle sunlight or with back lighting also has its problems, but these factors can be minimized by polarized filters on the crews' visors along with the use of artificial lights.

8.3.3.2 Repair

Due to the nature of the terrain and workload on the rover, there are bound to be some damage to the vehicle. Generally equipment failure can be rectified by switching to redundant systems. When this option is not available, on-site repairs may be attempted depending on the nature and extent of the failure. Some spare parts and a repair tools kit are provided with the rover for minor repairs. The kind of failures the astronauts will be able to fix are limited to replacing deformed wheels, broken lights and minor structural problems. A guideline will be set as to how the astronauts can quickly check through the systems to ascertain whether a certain breakdown is recoverable or not. In case of a irreparable damage it is advisable to switch on to emergency procedure, abandon the vehicle and be head back ton foot at the earliest.

8.4 Regolith Collector

8.4.1 Requirements

In order to provide a radiation protection layer on the habitat, the precursor mission needs to carry along an vehicle to handle regolith. The vehicle has to have minimum mass, volume and power requirement. The design of the vehicle should prefferably be multipurpose, as in not specific to the habitat regolith layer.

8.4.2 Trade Studies

There are basically two ways of handling the lunar soil. One option is to dig deep into the lunar soil, may be use explosives to first loosen up the soil and then scoop out the soil and dump it on the regolith support structure. This would require a bulldozer type of equipment. Our first cut mass estimate on a light bulldozer is 3000 kg and it takes up a volume of about 25 m^3 .

The other option is to scrape up the top layer of loose lunar soil using a brush and then either put it in bags or collect it in a container and lay it on the support structure. Mass estimate on a bagger type equipment and a conveyer is approximately 2000 kg. Total stowed volume is comparable to that of the bulldozer.

Other than a lower mass, the collector and conveyer combination is overall a simpler system. The ratio of human supervision to their work hour is less than that of the bulldozer which requires continuous manual control. Also, design a light-weight bulldozer is kind of a wasted design because higher dry mass of such a vehicle is generally considered a favorable factor. We believe a stronger and more massive bulldozer will be essential for future expansion of the lunar base. Hence for the first Columbiad mission, we decided to take a regolith collector.

8.4.3 Design

The general layout of this apparatus is based on a design of a lunar regolith bagging system created by students at the Georgia Institute of Technology in March, 1990. (See Figure 8-26).

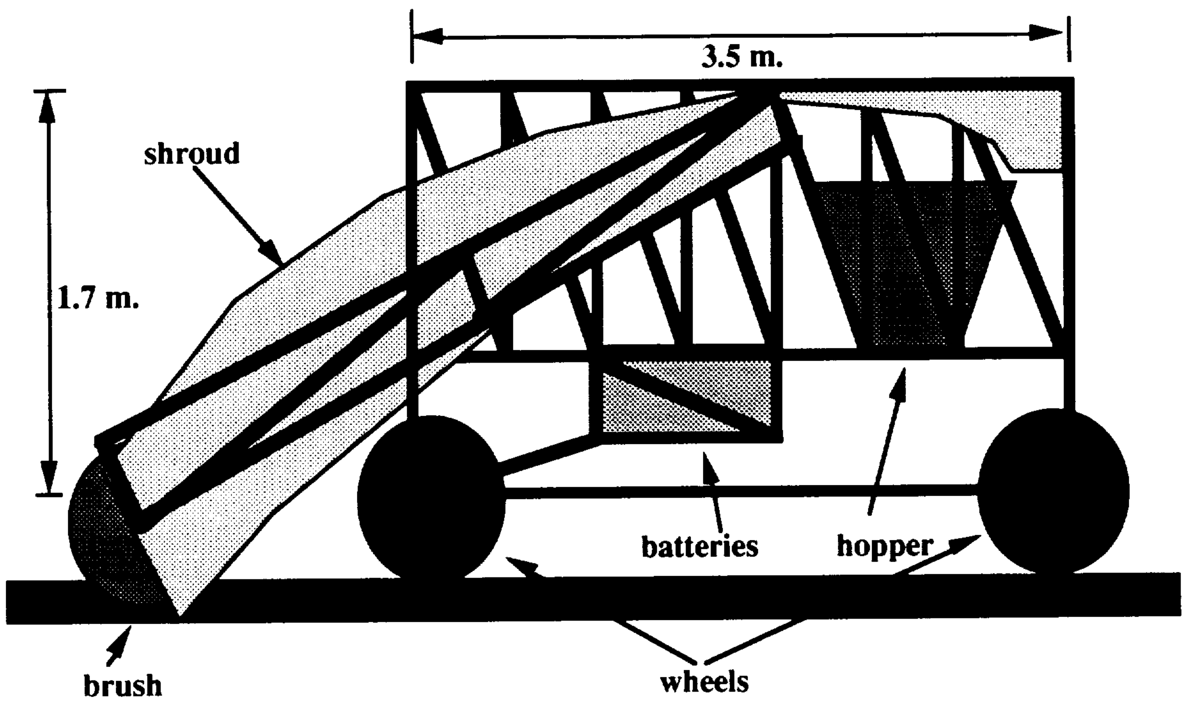


Figure 8-26
Side View of Regolith Collector

.4.3.1 Internal Structure and Layout

8.4.3.1.1 Sweeper Mechanism

The apparatus collects soil via a rotating brush., similar to a conventional street-sweeper. The specifications for the brush are a width of 1.6 m., a diameter of 0.6m, and a mass of approximately 2.82 kg. In order to survive the harsh radiation conditions in space, the material is modified from conventional polypropylene plastic to an aluminum/boron fiber composite.

In addition, this brush is rated for 150 RPM and requires 3.76 kW of power for operation. In order to fulfill this requirement, an electric D.C motor (2.27 kg) which supplies the needed power at 1750 RPM is selected. An 11:1 gear reduction assembly is therefore necessary. The motor and gear assembly is located at one end of the brush and enclosed in a protective case to avoid atmospheric exposure.

8.4.3.1.2 Shroud Assembly

Due to the lack of air resistance on the Moon, the trajectory of the thrown soil is easy to calculate - the regolith is lifted at an angle of 30 degrees from the surface. With this value, it follows that the soil would require a horizontal path of 5.6 m to lift up to the top of the hopper (1.6m). To maintain stability, a more compact design is desireable.

In order to shorten the apparatus to a total length of 4.3 m., the trajectory of the thrown regolith is guided by a casing. This shroud (183.00 kg) is constructed of 2.00 mm. thick box of boron/aluminum composite to guard against penetration by small rocks and has a interior coated with Teflon to provide a virtually frictionless path for the soil. In addition, the shroud protects the batteries, control modules, and other components from exposure to the high velocity soil particles.

The shroud is divided into two parts; most of its mass is attached to the brush armature truss in order to insure maximum capture of swept soil The smaller component is attached to the main body at two points and helps guide all stray soil into the hopper.

8.4.3.1.3 Hopper

A wedge-shaped aluminum hopper with a top cross-section of 1.1 m X 2 m. and a bottom cross-section of 0.5 m X 2 m. (approximately 25 kg) is placed beneath the lip of the shroud to collect up to 1.0 cubic meter of regolith. At the bottom, there are two hinged doors which open to allow the filling of the soil transport module of the rover when it is in position.

8.4.3.1.4 Winch Assembly

When the brush encounters compact soil or a rock, there will be great resistance to rotation. The control system is designed to respond to corresponding fluctuations in current and to inform the winch mechanism to raise the arm. Two thin cables, each experiencing a maximum tensile load of 98.17 N, are attached to the lower end of the brush armature, pass over pulleys at the front end of the main body, and are wound up on a spool located 1.0 m. from this front end.

8.4.3.1.5 Wheel Assemblies

The two front wheels of the regolith collector each have an assembly of two motors. One motor is provided for forward and backward movement while the second turns the entire wheel laterally. Masses of these motors are similar to the brush motor, i.e 4.45 kg each. Combined with a simple on-board computer, this setup allows the vehicle to be programmed to follow any path.

The wheels themselves have a 0.45 m. tread, 0.6 m. diameter, and are of similar construction to those used in the Apollo rover. They are comprised of spring-steel wire mesh carrying treads of titanium-alloy chevrons for traction.

8.4.3.2 Support Structure

8.4.3.2.1 Design Requirements

Due to spacecraft volume limitations, the soil collector is stored in the spacecraft unassembled and constructed on the lunar surface. In order to make this assembly easy during EVA, a simple truss design is implemented (See Figure 8-27).

Aluminum is selected for the struts because it can be threaded at the ends and used with screwable joints. Also, the geometry of the truss is massaged in order to reduce the number of discrete member lengths. The members are further generalized by sizing the cross-sectional area of each member to take the same maximum load. The result of all of these choices is a significantly reduced assembly time.

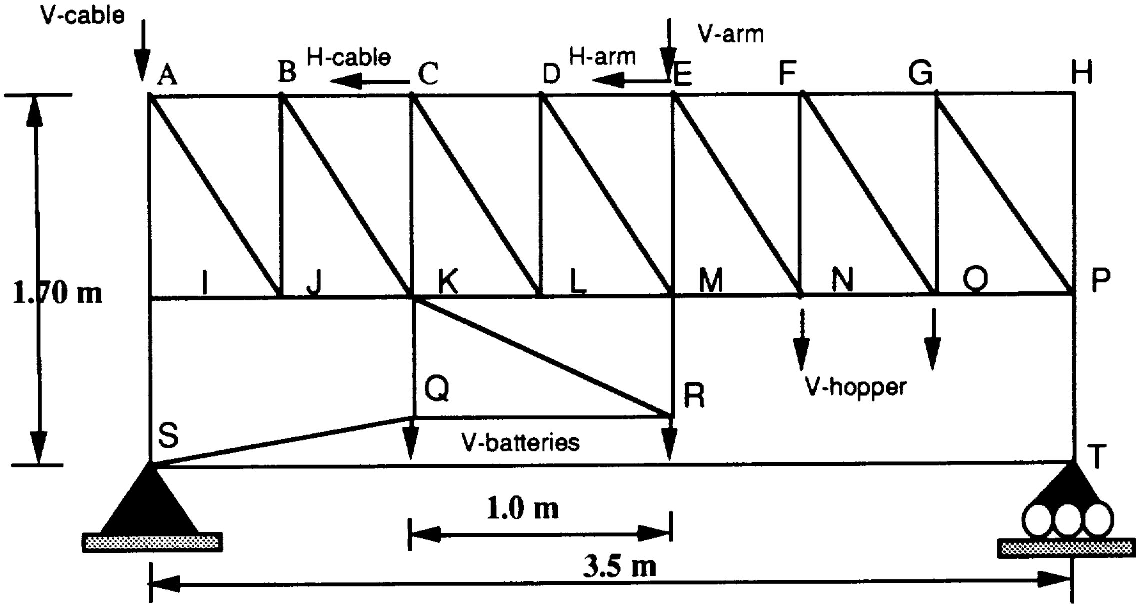


Figure 8-27
Main Body Truss

8.4.3.2.2 Analysis

In order to determine the largest strut loads, four distinct loading configurations, are considered. The two modes of operation of the collector are sweeping soil and moving to a new location. During the former (shown in Figure 8-28), the brush rests on the surface, contributing a vertical force due to the weight of the armature, as well as a horizontal friction force. The results of this analysis for both full and empty hopper cases are shown in Appendix II.

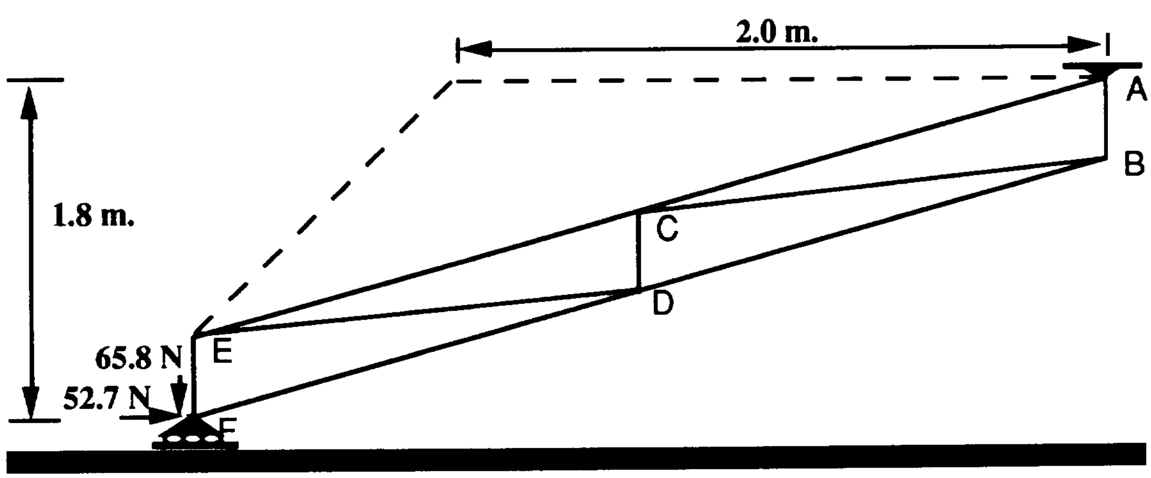


Figure 8-28
Armature Operation Mode 1-Sweeping Position

During the second mode of operation (shown in Figure 8-29), the armature is raised to a height of 0.5 m. above the lunar surface. This is done to allow for clearance of small obstacles. This is the mode where the load analysis is extremely important , especially when the hopper is empty, because the raised brush contributes a large moment on the system and can produce instability. In fact, the previous design of the main body failed in this case because the batteries were too far forward, causing the back end to be raised off the ground. The results of this analysis, again for both full and empty hopper cases are shown in Appendix II.

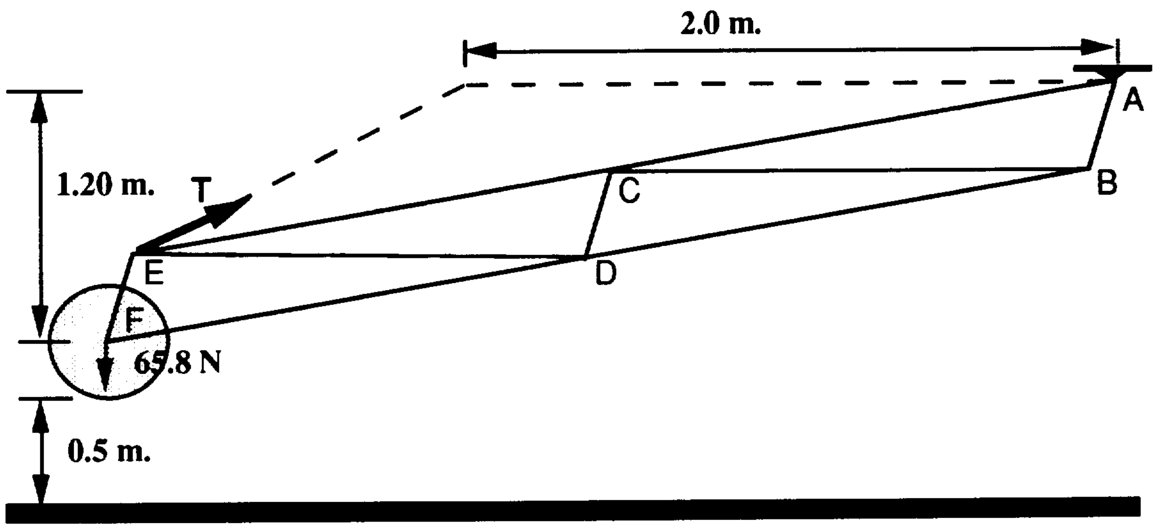


Figure 8-29
Armature Operation Mode 2-Arm Raised

The results of the analysis were as follows. The maximum load occurred while the collector was in the sweeping mode with a full hopper. The compressive load of approximately 1063 N set the moment of inertia to 1.90e-9 m4 to prevent buckling. The corresponding radius of a solid cross-section cylinder was 7.67 mm. Designing each of the members to this cross-section, a structural mass of 65 kg. is determined from a total member length of 91.72 m, including 13 - 2m cross members between side trusses.

8.4.3.3 Power System

The power requirement of the Regolith Collector is same as that of the Rover. Hence the on-board power systems are identical in design and performance specifications.

.c4.8.4.3.4. Overall Specifications

Table 8 -21: Overall Specifications of the Regolith Collector Design

Maximum Operating Time	8 hours
Recharge Time	12 hours
Power Consumption	6.5 kW
Dry Weight	614 kg
Collection Rate	3.5 m3 / hour

8.4.4 Operation

The following flowchart summarizes the operation of the Regolith Collector.

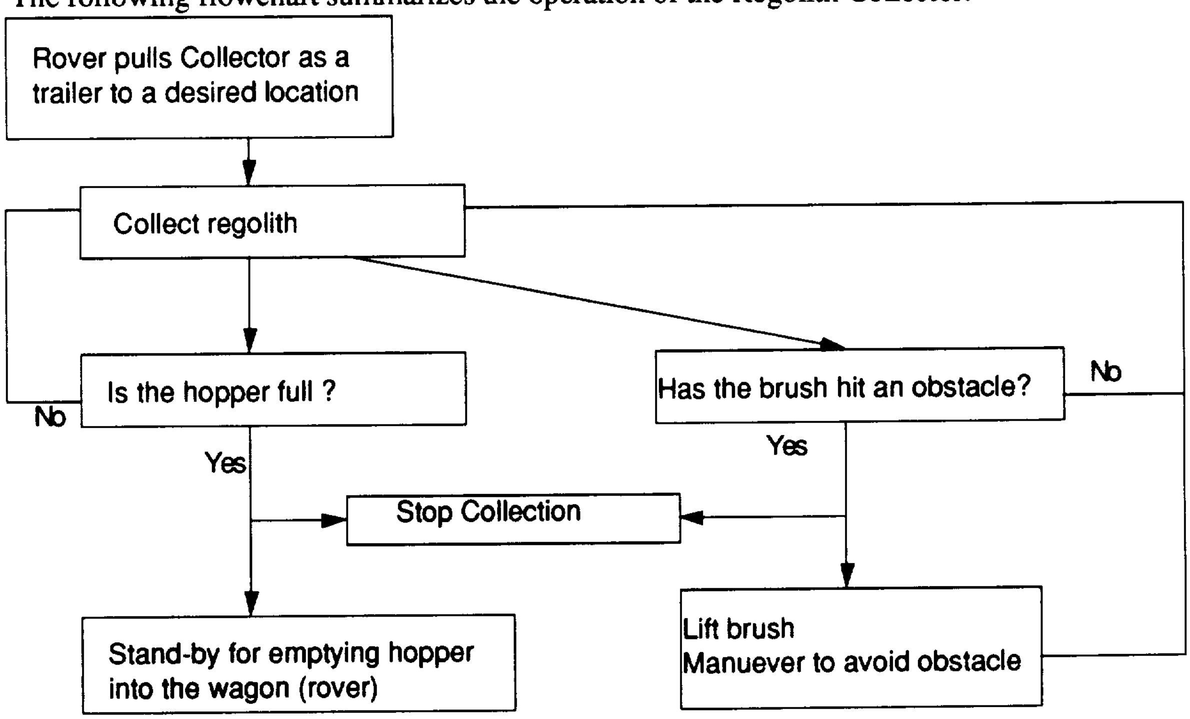


Figure 8-30
Operations flowchart for the Regolith Collector

8.5 Lunar Conveyer

8.5.1 Requirements

The Lunar Conveyer's primary duty is to transport regolith at different levels of the regolith support structure on the habitat. In order to work in conjunction with the Rover and the Regolith Collector, the Conveyer's dimensions must be compatible with the other vehicles. The Conveyer should be flexible to transport payload over different inclines. The flexible segments should have the provision of maintaining a rigid shape, in case the payload needs to be delivered across a trench.The mass and stowed volume of the entire system should be as low as possible.

8.5.2 Design

8.5.2.1 Description

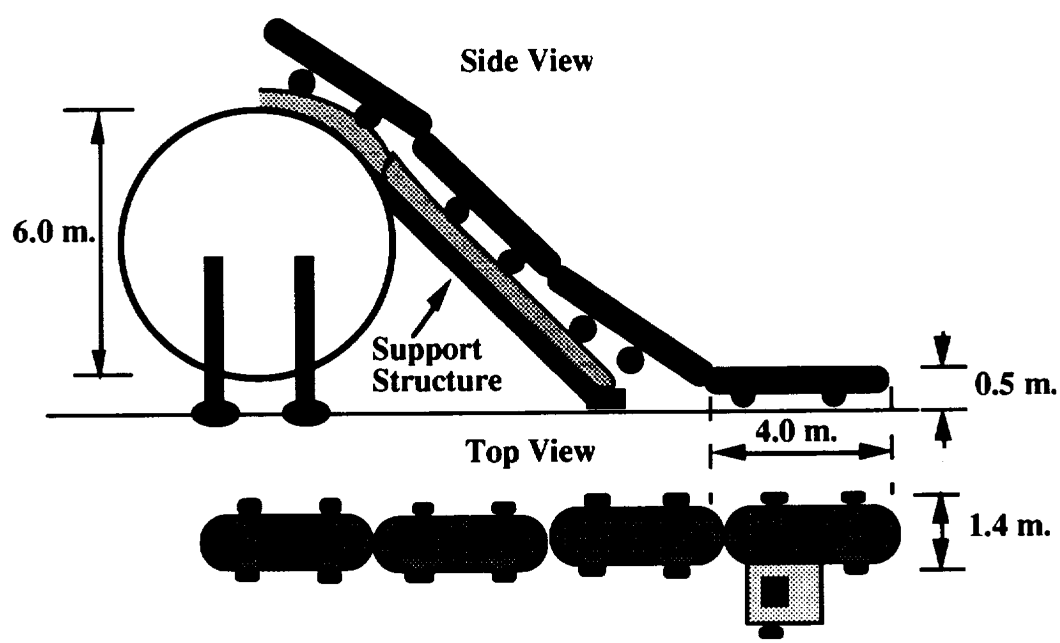


Figure 8-31
General View of Conveyor System

The conveyor system (Figure 8-31) consists of four segments connected together by pins to give the apparatus the flexibility to move up inclines, over ground obstacles, etc. This flexibility also allows the conveyor system to fold up for compact in-flight storage. Torsional clamps can be added at these connection points when performing maneuvers that require rigidity, i.e. bridging a hole. Each segment is 4.0 m. long, 1.4 m. wide (including tire treads), and 0.5 m. high.

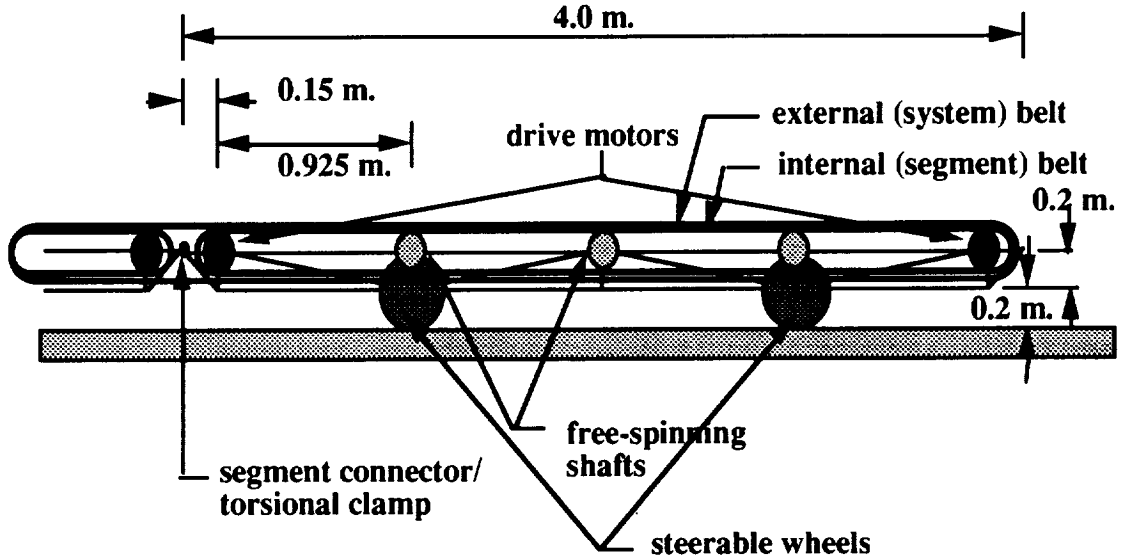


Figure 8-32
Diagram of Conveyor Segment

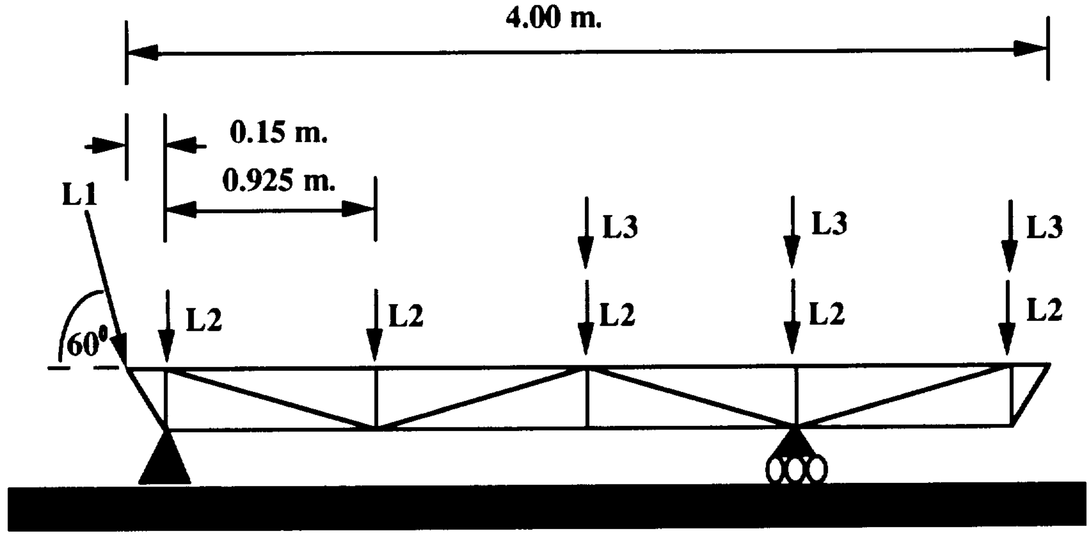
The segment design is illustrated in Figure 8-32. The segment belt system consists of two drive motors, one at each end, and three free-spinning shafts at the 1.0 m., 2.0 m., and 3.0 m. marks along the length of the segment. Each shaft has a radius of 0.08 m. The friction between the segment and system belts, insures maximum efficiency of each segment's drive motors.

Like the regolith collector, the wheels, located at the 1.0 m. and 3.0 m. points, are composed of spring-steel wire mesh carrying treads of titanium-alloy chevrons for traction. Each wheel has a 0.20 m. tread and a radius of 0.20 m. The former is necessary to support the worst case design loads (described in the next section). The latter is essential to give the conveyor belt enough clearance so that it will not touch the ground when the belt sags during maneuvers in a bent configuration.

The conveyor has 6-wheel drive: the front two wheels of the first segment and all four wheels of the wheel base are motorized. In addition, the front two wheels of the first segment are steerable to allow lateral motion along the length of the habitat.

8.5.2.2 Structure

A sketch of the truss one of the conveyor's four segments is shown in Figure 8-33. The segment consists primarily of two 3.7m X 0.2 m. box trusses held together by 1.0 m. horizontal crossmembers. The weight of the regolith is transferred from the drive motors and free-spinning shafts down through the truss to the wheels.



Loading Key:

L1 = 1600 N; weight of other three soil-laden segments on 60 degree slope.
L2 = 75 N; distributed weight of 0.38 cu. m. of regolith on segment.
L3 = 20 N; distributed weight of the controls of wheel base.

Figure 8-33
Segment Truss - Worst Load Case

In order to find the structural mass of this truss, the worst possible load case must be considered. This would be encountered when the wheel base segment is horizontal, experiencing the full weight of the loaded regolith, operator, batteries, and controls, while being loaded at the front by the first three segments resting on a 60 degree slope. A robust design is obtained by sizing all members to the cross-section corresponding to the maximum loaded member. In this case, the maximum load is roughly 3,034 N in compression in a 0.925 m. member. The corresponding radius of a solid aluminum cylinder is 9.02 mm. The totatl structural mass for the conveyor (8 2-D trusses and 44 cross-members) is estimated at 120 kg.

At each end of the segment, there is a rod which pins adjacent segments together while allowing them to rotate freely around the axis of the rod. This setup gives the conveyor the flexibility to deliver its payload up inclines and over obstacles, as well as the ability to be folded for compact storage. Enough clearance between flywheels (0.14m) has been provided to allow for the attachment of torsional clamps at these pins to hold the conveyor in a rigid configuration. This is useful when the conveyor needs to bridge gaps.

A wheel radius of 20 cm is necessary to give the conveyor belt enough clearance so that it will not touch the ground when the belt sags during manuevers in a bent configuration. Like the regolith collector, the conveyor's wheels will be made of spring-steel wire mesh with treads of titanium-alloy chevrons for traction and will require a tread of 0.20m in in order to handle the weight of the regolith.

8.5.2.3 Power

The Lunar conveyer requires 5 kW for running the conveyer, plus another 1.5 kW for running the vehicle itself. So for 8 hours of continuous operation it will need roughly 50 kWh or energy. Since the Lunar Conveyer is expected to work very close to the habitat, at least in this mission, it is not provided with on-board power supply. Instead, it is connected to SlurPP through a power umbilical directly. The option of carrying batteries and having the conveyer work on remote site remains in the design.

8.5.3 Operation

The Lunar Conveyer is driven up the 45° incline, on the regolith support structure. All the drive wheels are locked in position. The Rover wagon brings loose regolith from the Collecter and drives over the base of the Conveyer. The bottom of the wagon opens up and feeds regolith onto the conveyer. The conveyer carries the regolith to the top of the habitat and dumps it there. In order to control the accumulation of the regolith on any particular position on the habitat, the feed rate is varied. The maneuverability of the conveyer can also be used for some control. The conveyer can be moved to a different location, while maintaining a same height along the side of the habitat, by using the four-wheel drive capability of the vehicle.

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APPENDIX I

GROUP	Apparatus	TEMP	PRESSURE	SPECIAL
Crew Systems	Water Electrolysis System(2)	same as cabin	same as cabin	Built In;Continuous M
	Sabatier Apparatus	same as cabin	same as cabin	Has Oxygen Backup
	Crew Cabin	68F<T<81F	5 psi	
	Drinking Water Supply Tank(6)	same as cabin	same as cabin	Needs Continuous Volu
	Wash Water Supply Tanks(3)	same as cabin	same as cabin	Needs Continuous Vol
	Wash Water Filter	same as cabin	same as cabin	-
	Water Purity Indicator(4)	same as cabin	same as cabin	Used Only When Draw
	Air Quality Indicator(4)	same as cabin	same as cabin	Capsule & Habitat hav
		same as cabin	same as cabin	Toxin Content checke
	Particulate Filter System(5)	same as cabin	same as cabin	2 in capsule;3 in h
	Commode	same as cabin	same as cabin	1 in hab & caps;check
	Bio-Instrumentation(120)	same as cabin	same as cabin	Hab and Caps eac
	Radiation Detectors(6)	same as cabin	same as cabin	4 in Hab;2 in
				Rad Level< 25 RE
	EVA Suit(5)	75 F	.34 atm	Interior Temp ; Bl
				Oxygen Level;ide
				Suit Location Relati
	Bio-Inst for EVA Suit(20-80)	-	-	Checked Continuous
	IVA Suit(4)	75F	.34atm	Interior Temp;Bl
				Oxygen Level;i
	ECG(Multichannel)	-	-	Heart Electrical Acti
	Biobelt Assembly per EVA Suit	-	-	Heart Rate/Lung Volu
C3				
	HP GaAs Computer(3)	same as cabin	same as cabin	
	RH32 Data Processor	same as cabin	same as cabin	
	MDM-16 MUX/DEMUX	same as cabin	same as cabin	
	Odectics Tape OHSR	same as cabin	same as cabin	

	Fairchild Solid State	same as cabin	same as cabin	
	Univeral Demodulator(2)	same as cabin	same as cabin	
	High Data Rate Modem(2)	same as cabin	same as cabin	
	Antenna Pointing System	same as cabin	same as cabin	
	High gain antenna(2)	same as cabin	same as cabin	
	Low gain antenna(2)	same as cabin	same as cabin	
	Receiver(4)	same as cabin	same as cabin	
	Transmitter(4)	same as cabin	same as cabin	
	Power Supply HP	same as cabin	same as cabin	
GNC	Fiber Optice Gyroscope	same as cabin	same as cabin	
	accelerometer	same as cabin	same as cabin	
	sunsensors (2)	same as cabin	same as cabin	output-are starts in p
	earthsensors(2)	same as cabin	same as cabin	output-is earth in rig
	GPS receivers (2)	-	-	output
	Lunar landing Radars			
	altimeters	-	-	
	doppler	-	-	
	displays	-	-	
	docking system	-	-	
	lunar beacons	-	-	
PTC	Fuel Cells	yes	yes	
	Transformers	yes	yes	

APPENDIX II

Table 1 : Main Body Truss Analysis
Sweeping Mode / Full Hopper

Load Case 1A

External F's

Joint Name	Horizontal Force (in N)	Vertical Force (in N)	H moment arm (in m.)	V moment arm (in m.)	Moment (in N-m.)
A	0.00	0.00	0.00	0.00	0.00
C	0.00	0.00	0.00	0.00	0.00
E	-52.70	-33.80	2.00	-1.70	21.99
N	0.00	-500.00	2.50	0.00	-1250.00
O	0.00	-500.00	3.00	0.00	-1500.00
Q	0.00	-200.00	1.00	0.00	-200.00
R	0.00	-200.00	2.00	0.00	-400.00
Totals	-52.70	-1433.80			-3328.01

Reaction F's

Joint Name	Horizontal Force (in N)	Vertical Force (in N)
S	52.70	482.94
T	0.00	950.86

Member F's

Member Name	Force (in N)	Member Name	Force (in N)	Member Name	Force (in N)
AB	-236.25	FG	-475.54	MR	173.64
AI	-472.40	FN	450.86	NO	701.02
AJ	528.18	FO	-504.10	OP	475.54
BC	-472.50			HP	0.00

BJ	-472.40	GO	950.86	QR	-52.70
BK	528.18	GP	-1063.14	QS	-53.74
CD	-600.82	GH	-0.01	PT	-950.86
CK	-256.58	IJ	0.00		
CL	286.88	IS	-472.40	Checks	
DE	-729.14	JK	236.25	HP	0.00
DL	-256.58	KL	419.80	GH	-0.01
DM	286.88	KQ	189.46		
EF	-701.02	KR	58.92		
EM	-82.94	LM	548.12		
EN	54.94	MN	676.44		

Max. Force = -1063.14 N in member GP

Table 2 : Main Body Truss Analysis
Sweeping Mode / Empty Hopper

Load Case 1B

External F's

Joint Name	Horizontal Force (in N)	Vertical Force (in N)	H moment arm (in m.)	V moment arm (in m.)	Moment (in N-m.)
A	0.00	0.00	0.00	0.00	0.00
C	0.00	0.00	0.00	0.00	0.00
E	-52.70	-33.80	2.00	-1.70	21.99
N	0.00	-10.00	2.50	0.00	-25.00
O	0.00	-10.00	3.00	0.00	-30.00
Q	0.00	-200.00	1.00	0.00	-200.00
R	0.00	-200.00	2.00	0.00	-400.00
Totals	-52.70	-453.80			-633.01

Reaction F's

Joint Name	Horizontal Force (in N)	Vertical Force (in N)
S	52.70	272.94
T	0.00	180.86

Member F's

Member Name	Force (in N)	Member Name	Force (in N)	Member Name	Force (in N)
AB	-131.23	FG	-90.46	MR	173.64
AI	-262.40	FN	170.86	NO	175.90
AJ	293.38	FO	-191.04	OP	90.46
BC	-262.46			HP	0.00
BJ	-262.40	GO	180.86	OR	-52.70
BK	293.38	GP	-202.22	OS	-53.74

CD	-285.75	GH	-0.01	PT	-180.86
CK	-46.58	IJ	0.00		
CL	52.09	IS	-262.40	Checks	
DE	-309.05	JK	131.23	HP	0.00
DL	-46.58	KL	209.76	GH	-0.01
DM	52.09	KQ	189.46		
EF	-175.90	KR	58.92		
EM	127.06	LM	233.05		
EN	-179.85	MN	256.35		

Max. Force = -309.05 N in Member DE

Table 3 : Main Body Truss Analysis
Arm Raised / Full Hopper

Load Case 2A

External F's

Joint Name	Horizontal Force (in N)	Vertical Force (in N)	H moment arm (in m.)	V moment arm (in m.)	Moment (in N-m.)
A	0.00	-98.17	0.00	0.00	0.00
C	-98.17	0.00	0.00	-1.70	166.89
E	-53.78	-16.33	2.00	-1.70	58.77
N	0.00	-500.00	2.50	0.00	-1250.00
O	0.00	-500.00	3.00	0.00	-1500.00
Q	0.00	-200.00	1.00	0.00	-200.00
R	0.00	-200.00	2.00	0.00	-400.00
Totals	-151.95	-1514.50			-3124.35

Reaction F's

Joint Name	Horizontal Force (in N)	Vertical Force (in N)
S	151.95	621.83
T	0.00	892.67

Member F's

Member Name	Force (in N)	Member Name	Force (in N)	Member Name	Force (in N)
AB	-246.69	FG	-446.45	MR	124.01
AI	-591.44	FN	392.67	NO	642.82
AJ	551.52	FO	-439.04	OP	446.45
BC	-493.38			HP	0.00
BJ	-493.27	GO	892.67	QR	-151.95
BK	551.52	GP	-998.08	QS	-154.96

CD	-519.07	GH	-0.02	PT	-892.67
CK	-247.67	IJ	0.00		
CL	276.91	IS	-591.44	Checks	
DE	-642.93	JK	246.69	HP	0.00
DL	-247.67	KL	341.43	GH	-0.02
DM	276.91	KQ	169.61		
EF	-642.82	KR	169.89		
EM	-123.66	LM	465.29		
EN	120.00	MN	589.15		

Max. Force = -998.08 N in Member GP

Table 4 : Main Body Truss Analysis
Arm Raised / Empty Hopper

Load Case 2B

External F's

Joint Name	Horizontal Force (in N)	Vertical Force (in N)	H moment arm (in m.)	V moment arm (in m.)	Moment (in N-m.)
A	0.00	-98.17	0.00	0.00	0.00
C	-98.17	0.00	0.00	-1.70	166.89
E	-53.78	-16.33	2.00	-1.70	58.77
N	0.00	-10.00	2.50	0.00	-25.00
O	0.00	-10.00	3.00	0.00	-30.00
Q	0.00	-200.00	1.00	0.00	-200.00
R	0.00	-200.00	2.00	0.00	-400.00
Totals	-151.95	-534.50			-429.35

Reaction F's

Joint Name	Horizontal Force (in N)	Vertical Force (in N)
S	151.95	411.83
T	0.00	122.67

Member F's

Member Name	Force (in N)	Member Name	Force (in N)	Member Name	Force (in N)
AB	-141.67	FG	-61.36	MR	124.01
AI	-381.44	FN	112.67	NO	117.71
AJ	316.72	FO	-125.97	OP	61.36
BC	-283.33			HP	0.00
BJ	-283.27	GO	122.67	QR	-151.95
BK	316.72	GP	-137.16	QS	-154.96

CD	-204.00	GH	-0.02	PT	-122.67
CK	-37.67	IJ	0.00		
CL	42.12	IS	-381.44	Checks	
DE	-222.84	JK	141.67	HP	0.00
DL	-37.67	KL	131.38	GH	-0.02
DM	42.12	KQ	169.61		
EF	-117.71	KR	169.89		
EM	86.34	LM	150.22		
EN	-114.79	MN	169.06		

Max. Force = -381.44 N in Member AI

VOLUME IV
Program Plan

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1 Program Plan - Philosophy

This chapter will address the detailed philosophies behind the Columbiad Program Plan. Items discussed will include the purpose and phases of the program, the content and design process for the program, and descriptions of the proposed management, technical, procurement, and resources approaches for the program.

The Columbiad Program will consist of a collection of staffed and unstaffed elements to be developed in an integrated program. The following definitions have been established for the Columbiad Program. (A more detailed Columbiad Program lexicon will be prepared at a later date. When approved, it will be a controlled document and will contain all appropriate Columbiad definitions).

Columbiad Program: The collection of manned and unmanned projects to be developed as an integrated program leading to the return of a human presence on the Moon by the year 2000. The Program includes the reasearch, development, testing, and evaluation (RDT&E) period as well as the operation of the five year campaign.

Columbiad Campaign: The five year opertional fight time for the program. The Columbiad Campaign will begin with the first operational flight, projected for the year 2000. The campaign will include all flight missions with all necessary support and management operations.

Piloted Mission: Any mission in which a human crew is launched from the Earth with the intent of landing on the lunar surface. This mission includes the crew, the Crew Module (CM), the Earth Return Module (ERM), the Lunar Braking Module (LBM), and the Primary Trans-Lunar Injection stage (PTLI). Two launches of the National Launch System (NLS) are required to place these modules and stages into low Earth orbit (LEO).

Precursor Mission: Any mission in which material and equipment is launched as a payload from the Earth with the intent of being landed and deployed on the lunar surface. The first of these missions will precede the first piloted missions. Subsequent precursor missions will be launched as scheduled and/or needed. This mission includes the payload, the Payload Landing Module (PLM), the LBM, the PTLI, and two NLS vehicles. Two launches are required to place the payload and modules into LEO.

Module: An attachable/detachable Columbiad element that provides a unique or common function for a mission.

1.1 Purpose of Plan

The purpose of this Program Plan is to describe the overall technical, management, and procurement approaches for the Columbiad Program. The plan is structured so that it summarizes the activities entailed in implementing the program. It also describes the technical and management plans, procurement strategy, schedules, and resources required for implementation. The primary emphasis at this time is on the definition and preliminary design phase and the planning and definition of issues associated with progressing into the program development phase. As the Columbiad Program matures, this document will be revised to provide greater detail on program development and follow-on phases of the program.

1.2 Program Phases

The top-level planning schedule for Project Columbiad is shown in Figure 1-1. All planning activities are geared towards an initial operational capability (IOC) by the year 2000. The schedule includes activities to support the five year campaign that follows the IOC. Details are discussed in Chapter 3 of this volume

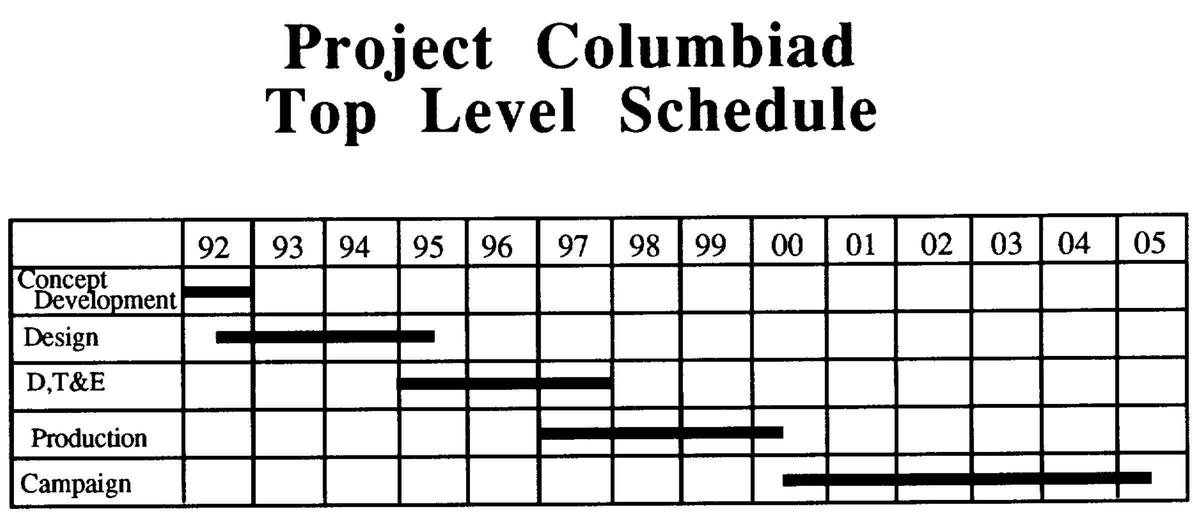

Figure 1-1
Columbiad Program Overall Top-Level Schedule

1.2.1 Definition Phase

Because Project Columbiad is a long-term effort, it is essential that corporate memory be retained within a single entity. Thus, Hunsaker Aerospace Corporation (HAC) has performed the systems integration and engineering tasks in-house for this preliminary design; and it is being transferred at this time to NASA (structure for NASA's management is discussed in a later section).

The program definition phase, including the preliminary design, has been conducted and is presented in the body of this report. This definition phase provides the preliminary design for an IOC by the year 2000 as well as the design for the five year operational campaign. Through the definition phase, HAC has defined the program mission, defined the operations and systems requirements, performed trade studies, designed support systems, developed a preliminary system design, defined system interfaces, developed cost and schedule estimates, and prepared detailed plans for the development phase. Many of these accomplishments have already been presented in this report. Items concerning cost, scheduling, and development phase plans will be discussed in this volume.

1.2.2 Development Phase

The purpose of the development phase is to design, manufacture, integrate, verify, test, and deliver the elements of the Columbiad Program with an IOC by the year 2000. The initiation of the development phase is scheduled to begin as soon as the review of this preliminary design is complete. The major divisions in this phase, as shown in Figure 1-1 above, include final program design, systems development, testing, and evaluation (D, T & E), module production, flight testing, and the five year campaign (starting with the first mission in the year 2000).

1.3 Program Goals and Objectives

The overall goals and objectives of the Columbiad Program have been presented in detail in Volume I of this report. They are summarized here to support the design and management approaches that will be proposed later in this volume.

The following Primary Program Objectives compose the foundation of the Columbiad Program:

- Transport a minimum of four people to the Moon and back.
    - Land at any latitude
    - Mission duration of 14 to 28 days
- Establish a foundation for a lunar base.

Secondary Program Objectives include the following:

- The establishment of a stepping stone for a piloted mission to Mars.
- Provide scientific research and exploration on the Moon.

- The establishment of international cooperation for space exploration.
- A boosting of national confidence in the arena of space flight and space operations.

An overall goal of the Columbiad Program is to meet the program objectives in the most efficient manner with respect to cost and scheduling.

1.4 Preliminary Design Process

This section covers the design process from the formation of the Columbiad Program team to the definition of mission requirements which need to be met by every subsystem group. The purpose is to provide an efficient corporate and communications framework so that mission goals can be accomplished in the limited period. First, a brief description of the corporate architecture is outlined, followed by the work breakdown structure (WBS) of each group. Then, a requirements tree tracing the top-level requirements to every mission stage requirement is described. Finally, reliability requirements are set for every level of the mission.

1.4.1 Corporate Architecture

The vast scope of the Columbiad Program required a detailed top-down corporate structure to provide an efficient and traceable communications link between all groups. Even though the small size of The Hunsacker Corporation allowed an informal flow of ideas from every level of the team from the advisory board to the subsystems groups, a hierarchical responsibilities network needed to be established.

The Corporation Structure (see Figure 1-2) is divided up into the advisory board, the planning and organizing groups, the project groups, and the subsystems groups. The purpose and tasks of each group is expanded in the WBS section below. For purposes of communication traceability, however, a slightly different structure was needed. The Communication Structure (see Figure 1-3) is again divided in a hierarchical fashion, but this time levels indicate the amount of integration which is needed for the project. For example, even though Status is a subsystems group, their work is directly affected by any integration necessary in the projects groups, therefore, this group needs to be on the same communications level. The solid lines indicate the vertical communication levels while the dotted lines trace the horizontal communications.

1.4.2 Work Breakdown Structure

Every level of the team has responsibilities important for the completion of this project. Starting from the planning and organizational groups to the subsystems groups, work breakdown structures (WBS) are established to ensure that all levels of the Columbiad Program are covered, from technical designs and interface integration to program planning and cost analyses.

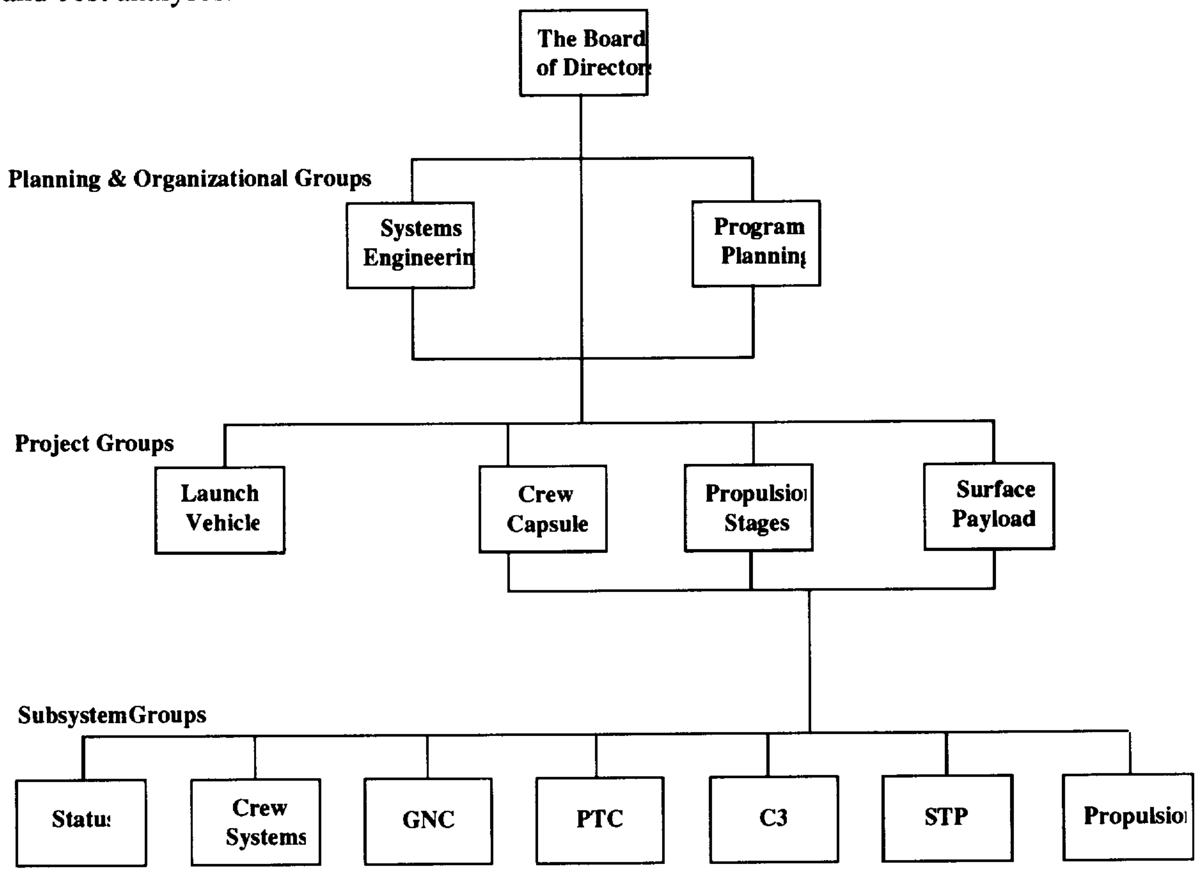

Figure 1-2
Corporation Structure

1.4.2.1 Planning and Organizational Groups

Systems Engineering and Program Planning are established to plan and organize the design process and the program process, respectively. Systems' responsibilities include performing mode studies and trajectory analyses to determine a detailed mission profile, developing quantitative system specifications and defining trade studies for each of the mission stages and project subsystems, and defining the surface mission. Other administrative tasks to promote traceability from overall system to subsystem specifications by developing a work breakdown structure, requirements tree, communications tree and reliability tree were also the responsibility of Systems.

Program Planning was activated during the second half of the design process to develop detailed cost analyses, develop the overall program philosophy and schedule, define the qualification and acceptance test programs, and to estimate reliability and maintainability of the every subsystem.

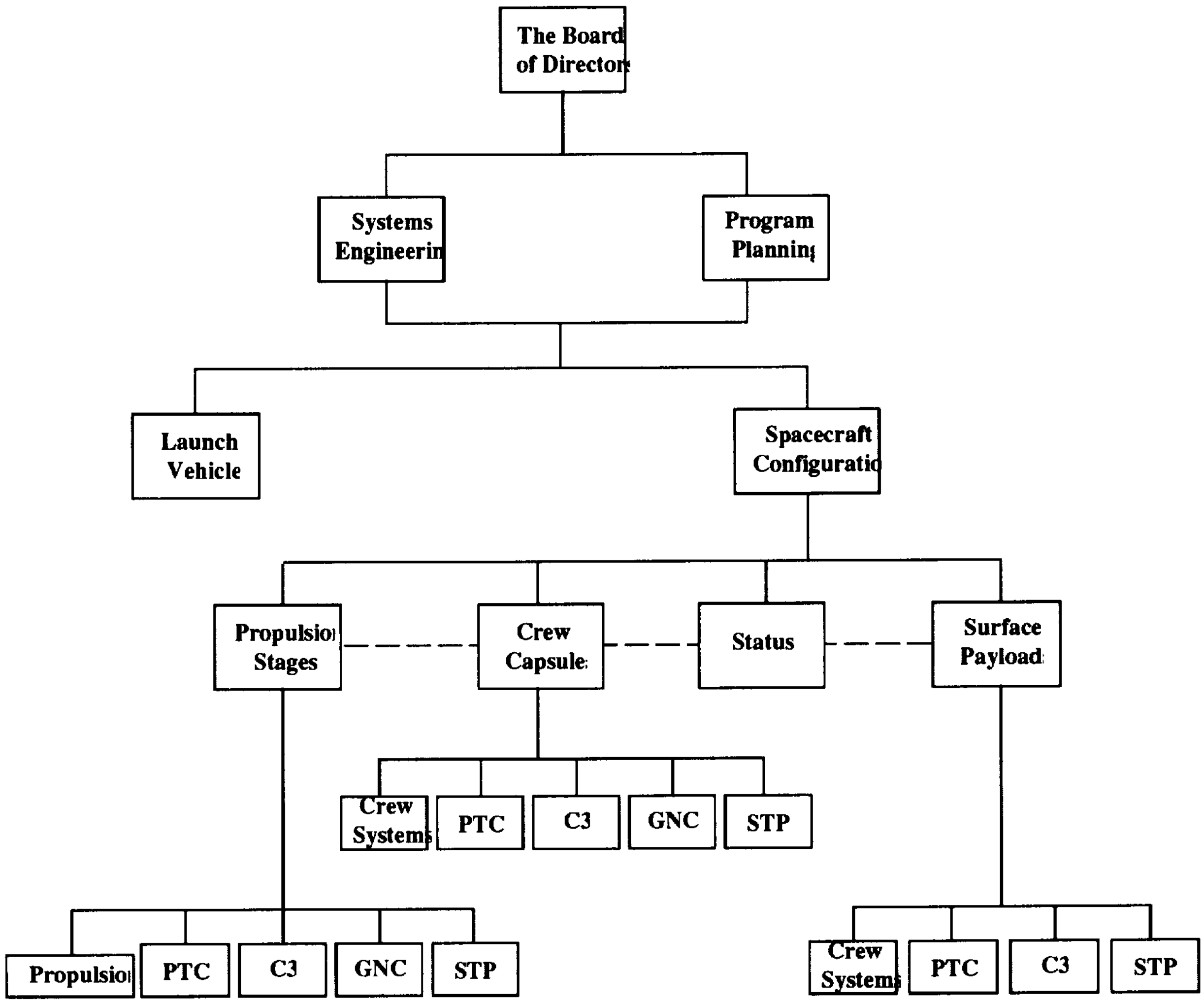

Figure 1-3
Communications Structure

1.4.2.2 Project Groups

Project groups were established to oversee and integrate all the subsystem components in each phase of the mission, from Launch Vehicles and Propulsion Stages to Crew Capsules and Surface Payloads. Launch Vehicles are responsible for considering the spectrum of possible vehicles, defining launch site capabilities, and designing any launch site changes.

Propulsion Stages' main task is developing staging configurations for the mission. This includes developing and maintaining budgets and margins for weight, propellant, power, guidance and navigation, control, and communications as well as integrating all these factors into detailed Interface Control Documents (ICD).

Crew Capsules are involved with the development of all the components pertaining to the Crew Module. This group determines where all subsystem components fit into the CM, develops and maintains budget and margins on weight, power, computer and communications resources, and design ICDs.

Surface Payloads are responsible for the design of the habitat, rover(s), and landing configuration. This group also integrates all the power, communications, and structures for the surface mission. A major design responsibility is to insure that astronauts are protected from radiation and certain levels of solar flares.

1.4.2.3 Subsystem Groups

Subsystem groups are responsible for the actual detailed designs for the project. These are divided up into seven groups: crew systems, guidance and navigation control, power and thermal control, command communications and control, structures and thermal protection, propulsion, and status. All groups are required to minimize cost, mass, and power in their subsystem components.

Crew Systems (CS) design all human life support equipment for the capsule and the surface mission. This group must define acceptable environmental conditions for the astronauts, incorporate appropriate consumables and recycling/disposal systems into the habitat, and consider health effects due to zero and one-sixth earth g gravity.

Guidance and Navigation Control (GNC) is responsible for the guidance and navigation of the spacecraft in its flight to the moon and back. These include systems that provide redundancy, midcourse corrections, and the capability for anytime abort. Landing, displays, rendezvous, pre-deployed navigation aids are factors that need to be considered.

Power and Thermal Control (PTC) provides primary and secondary power for all stages and the surface mission. Power is required for all life support in the capsule and habitat to provide thermal control during all phases of the mission, and to operate all GNC and communications systems.

Command, Communications and Control (C3) is responsible for all voice, data, and video communication links between the earth and spacecraft, earth and surface operations, and all components of the surface payload. Designs include on-board computers for data storage of status information to crew and earth, and for running guidance codes.

Structures and Thermal Protection (STP) designs the structural shell for all propulsion stages and the crew capsule. Not only does this group consider the loading forces encountered by the spacecraft during all phases of the mission, it also must account for thermal stresses.

Propulsion (prop) is responsible for the primary and reaction control propulsion systems. Given the high survivability requirement, this group also considers all the abort systems in the mission.

Status is responsible for the testing and evaluation qualifications for the spacecraft in its development and production phases as well as the vehicle health during the mission. This group provides the means to monitor vehicle and crew health during the mission, devises the conditions for which abort is necessary, and determines the manpower needed to handle ground operations.

1.5 Proposed Management Approach

This section describes the program management structure, participants and their responsibities, and the program control system. Procedures for direction, review and reporting, documentation, and information management are also included. This section looks beyond the HAC structure and philosophy that guided this preliminary design and proposes the management methods needed by NASA to carry this program to successful completion.

1.5.1 Program Participation

The Columbiad Program will be a national commitment; one that will involve NASA, other US government agencies and departments, and private commercial contractors. A concerted effort must be made in the conclusion of the definition phase to identify these participants and to define their degree of involvement. The following paragraphs summarize potential participation in the Columbiad Program.

1.5.1.1 National Aeronautics and Space Administration (NASA)

NASA is the responsible government agency for managing and directing all aspects of the Columbiad Program, including the definition of requirements, the program definition, design and development, and operations. NASA will work directly with the various US government agencies and departments in defining their respective involvements and/or requirements. As appropriate, memoranda of understanding and interagency agreements will be developed with participants to define roles, responsibilities, financial arrangements, and management relationships.

1.5.1.2 Department of Defense (DOD)

Because the Columbiad Program is a national program, all government agencies will be permitted a degree of involvement, including the DOD. Any involvement by any agency will be at the discretion of NASA. At present, the DOD has identified no requirements for militray personnel or equipment for the Columbiad Program. Therefore, the program is being structured on the basis of civil requirments. It is conceivable, however, that in the future there may be DOD involvement with either the piloted missions or the precursor missions, or both.

1.5.1.3 Other Government Agencies

Other government agencies may be involved in the Columbiad Program. Discussions will be held with interested agencies to define their potential involvement in the program.

1.5.2 Program Management

This section discusses possible management structures for the Columbiad Program.

1.5.2.1 NASA Headquarters

NASA will serve as the headquarters for the management of the Columbiad Program. It will house the Columbiad Program Office (the first level of management) and oversee its relationship with the NASA Field Centers (the next level of management). In order to establish agreed upon roles and responsibilites and distribution of funds between the Headquarters and the Field Centers, formal agreements will have to be met. These will be in the form of program and project initiation agreements established at the various management levels.

1.5.2.1.1 Columbiad Program Office (Level A)

The Columbiad Program Office will be established at NASA Headquarters and will be responsible for establishing program policy, budget and schedule guidlines, and for coordinating and interfacing with all external elements. This office will also be responsible for providing program direction and management, program requirements definition, control, utilization and operations planning and implementation, programmatic planning and control, and advanced program planning.

Due to the desire to streamline the management process and promote efficiency throughout the entire program, only this single Level A management approach will be utilized. Thus, activities usually undertaken by a Level B management center will be assumed by divisions of the Columbiad Program Office (Level A). These include the following:

Systems Engineering and Integration: Establish and manage the technical content of the Columbiad Program , in response to the established system requirements.

Business Management: Manage the program resources to the budget and schedule guidelines provided by the program directors.

Operations Integration: Assure that the Columbiad Program operations considerations are properly incorporated in the derivation of requirements and design of the system.

Selt-Support of Level A: Provide overall support for Level A activities during budget and schedule formulation, establishment of system requirements, and other aspects of program direction.

1.5.2.1.2 Additional NASA Offices

The following additional offices will be involved in the development and operation of the Columbiad Program. An exact definition of their degree and level of involvement is to be determined.

The Office of Space Flight (OSF) will be involved with the NLS and other lanuch systems, if any, used for the flight test program. OSF will also have some involvement in other aspects of each mission, since space flight plays a major role. OSF will interface with the Columbiad Program Office on all transportation requirements for the program.

The Office of Space Science and Applications (OSSA) will be responsible for the establishment of science and application requirements for the program. OSSA will also be responsible for the definition, design, and development of science and application payloads for each mission throughout the five year campaign.

The Office of Aeronautics and Space Technology (OAST) will be responsible for the management and execution of generic technology and supporting studies applicable to the Columbiad Program. It is important that any focused technology tasks undertaken by OAST and any advanced development tasks initiated by the Columbiad Program Office be closely coordinated and integrated.

The Office of Space Tracking and Data Systems (OSTDS) will be responsible for planning, defining, and budgeting for communication tracking and data acquisition systems and networks. OSTDS will interface with the Columbiad Program Office in these areas.

1.6 Proposed Technical Approach

This section describes the overall technical approach and activities for conducting NASA Headquarters (Level A) tasks. These activities include the top level approaches for research, design, development, and integration of all elements of the Columbiad Program. Additionally, approaches for testing, ground support, and maintenance will be discussed.

1.6.1 Research and Development Philosophy

The activites related to the technical research and development of the Columbiad Program will be discussed in this section.

1.6.1.1 Engineering Activities

These activities include unique mission considerations, the systems engineering and integration function, and hardware commonality.

1.6.1.1.1 Unique Considerations

The Crew Module (CM) is designed to be the only reusable and refurbishable module of the Columbiad Program. This makes it unique in research, development, and design approach; there will be expenses related to the refurbishment process. However, the reusability of the CM makes it attractive for the long term planning of the Columbiad Campaign.

The requirements for launch to any point on the lunar surface and the need for continuous abort possibilities are also unique considerations to be taken into account in the research, design, and development of the program.

1.6.1.1.2 Systems Engineering and Integration (SE&I)

The SE&I efforts consist of tasks required to define and analyze elements, systems, and subsystems of the Columbiad Program. The Columbiad Program Office will be responsible for establishing and implementing an in-house SE&I capability.

The SE&I function is ultimately responsible for systems engineering and integration, programmatic activites, and products. For this preliminary design certain activities in this range have already been performed, but will need to be re-evaluated by the Columbiad Program Office for program continuity. These activities include system analysis, system trades, definition synthesis, configuration analysis, systems requirements, requirements integration, and Interface Control Documents (ICDs). Tasks not undertaken for this preliminary design that will be under the direction of the SE&I function include maintainablity, technical management, and logistics plans, as well as the development of detailed specifications.

1.6.1.1.3 Hardware Commonality

The Columbiad Program will incorporate hardware commonality to the maximum possible extent. The desired effect will be to minimize cost through significant cost avoidance, to simplify integration, maintenance, and spare requirements, and to provide compatibility among all elements. It is also desirable to have a specified degree of modularity, particulary for command, control, and communications and guidance & navigation subsystems, to lower cost and simplify mission and module integration.

1.6.1.2 Advanced Technolgy Development

The feaseablity of an IOC for the Columbiad Program by the year 2000 is largely based on the use of existing technologies. However, there are a few aspects of the program that can be considered "advanced technology" and will undergo a period of research and testing before applications to the program can be utilized. These aspects of the program include the large scale production of a manocoque structure, the aerodynamic performance of the biconic capsule design, and certain portions of the crew systems and the command, control, and communications subsystems.

The advanced technology required for the development and production of the NLS is not considered in the program plan for this preliminary design. It is assumed that this will be developed independently and purchased as a launch system to be utilized for this program.

The approach to this advanced technology development involves three elements: focused technology, prototype technology, and test beds. The focused technology activites will be directed to insure development of the technology and its direct application to the Columbiad Program. It is necessary that advocacy and funding be made available to continue focused technology development through demonstration at the laboratory level. The prototype technolgy activity will continue the development process into prototype components. These will then be transfered to test beds and integrated with the necessary minimal subsystems to allow direct tests of the advanced technology. This test bed approach can include both ground and flight testing with specific requirements to be determined. NASA Centers will implement and operate tests beds and will assure their availability and use in the testing and evaluation of these advanced technologies. However, integrated ground and flight testing of complete subsystems and modules will be handled at a program-wide level and their management will be directed by the Columbiad Program Office.

1.6.1.3 Safety, Reliability, and Quality Assurance (SR&QA)

The Columbiad Program is designed to be safe and reliable and is expected to be developed, produced, and operated with these factors enforced through a rigorous quality assurance program. It is imperative that the design, development, production, and operational requirements be met. It will be the responsibility of the Project Columbiad Program Office to direct the management of SR&QA activities.

The SR&QA approach for the Columbiad Program will be in conformance with NASA management instructions. Since one of the goals of the program is to reduce cost and promote efficiency in the design, development, production, and operation of space systems it is imperative that a complete SR&QA program be implemented. This will include the application of Total Quality Management (TQM) principles and the continuation or confirmation or various system and subsystem trade studies. A planned SR&QA Program would include:

A Safety Program implemented to assure that hazards inherent in the space operations of the Columbiad Program and its ground systems are identified. This program will establish controls to eliminate the hazards or minimize them by incorporating safety factors, safety devices, caution and warning devices, redundancy, backup systems, and/or abort and emergency procedures. Much of the preliminary design work in most of these areas has been completed for this report and can be found in their various sections.

A Reliability Program implemented to assure through various management, engineering, and test activities that all Columbiad Program hardware designs meet the program's objectives and performance requirements.

A Quality Assurance Program implemented to validate the acceptability and performance characteristics of materials, components, subsystems, systems, and modules. This will assure the detection and correction of all departures from the design and performance specifications during the design, development, production, and operation of the Columbiad Program. This QA Program, as well as all other aspects of the entire Columbiad Program, must be managed with TQM principles to insure immediate testing, evaluation, and correction of all specification deviations and support a unified, NASA and nation wide, approach towards the total quality of the program.

1.6.1.4 Environmental Impact Assessment

In order to comply with all Federal and International regulations and well as environmental common sense it will be necesary for NASA to completely assess the potential environmental impact of the implementation of the Columbiad Program in all of its phases. This could be done within NASA or by an outside agency, with the restriction that this activity, like all others, remains under the central management of the Columbiad Program Office at NASA Headquarters.

1.6.2 Testing and Evaluation Philosophy: From Design To IOC

The success of the Columbiad Program depends on the success of the systems and subsystems that compose the modules as well as the overall successful operation of all equipment. Throughout the design and development (including production) phases, up to the time of IOC, testing and evaluation will be conducted to support the SR&QA programs identified above. Management of the testing and evaluations programs will be directed by TQM principles and will be the responsiblility of the Columbiad Program Office.

1.6.2.1 Testing Considerations and Drivers

The testing considerations that drive the Columbiad Program are derived from the top level mission requirements discussed in Volume I of this report. The system reliability is 99.9% for human survival, requiring three levels of redundancy in related subsystems. The reliability for mission success is set at 95%, requiring two levels of redundancy in related systems and subsystems. These requirements must be considered in the design of specific tests and test programs (not covered in this preliminary design).

Trickling down these requirements to the subsystem levels identifies the design driving considerations. Crew systems is designed with a factor of safety (FOS) of 1.5. The propulsion systems are designed with a FOS of 1.1. The FOS for both structure and thermal protection subsystems are 1.4. The foundations for these considerations are found in Volume I of this report. The testing procedures to confirm the meeting of these design specifications are to be designed and carried out both on site at the contractor's facility and at the subsystem integration facility under the direction of the Columbiad Program Office.

1.6.2.2 Ground Testing Phases

This section discusses the phases of testing performed on the ground in support of the testing and SR&QA philosophies of the Columbiad Program. These are testing phases for the production aspects of the program, prior to the IOC. The prototype testing for new technologies is expected to be done under the direction of the advanced technology development program identified in section 1.6.1.2 of this volume. This section also describes the composition of current tests that could be utilized for this phase.

1.6.2.2.1 Component Qualification Testing

Before individual components are received from the supplier/contractor it is expected that test articles will have been previously evaluated with reference to the design specifications. These tests and evaluations shall consist of tests done at specified mission loads and separate tests performed at 1.5 times the mission loadings. These loadings can be categorized as static, dynamic, and thermal loadings.

The static load series will include inertia, applied and pressure loads. The dynamic loads series will include vibrational, acoustic loads, shock, and impact tests. The thermal loadings tests will be used to prove the flight worthiness of the component, subsystem, and module thermal protection system or thermal resistance (radiation shielding testing is included in this category).

All these tests will be performed on-site in adherence to standards enacted by the Columbiad Program Office. The testing results will be incorporated immediately into design alterations as necessary.

True and final component qualification testing will occur at the subsystem integration facility. This facility and program will be entirely controlled by the Columbiad Program Office. This program will test the components at the mission load levels before subsystem integration occurs. Verification of the 1.5 times mission loading tests must be obtained from the supplier/contractor before component qualification testing and integration can begin.

1.6.2.2.2 Subsystem Intergration Testing

This testing will occur at a NASA-controlled facility, as identified above. It will occur at various stages of subsystem integration, as identified by the Columbiad Program Office or its appointees. The final integration testing will occur after the entire subsystem, with its support systems, is integrated. The degree of testing beyond or in addition to mission loadings and specifications is to be determined. If integration of a subsystem requires partial or complete integration of a module, then these activities can be allowed to proceed simultaneously if the facility can support such activities. It thus is recommended that the module integration facilities contain subsystem integration facilities as well.

1.6.2.2.3 Module Assembly and Testing

The facilities for module assembly and testing will be NASA-controlled and administered by the Columbiad Program Office. As noted above, they may contain subsystem integration and testing facilities. Testing will be designed to determine that module design specifications have been met. Mock-ups for testing up to failure loads and for maximum environmental conditions testing will be provided for the facility as required and deemed necessary. Following module qualification it will undergo a final check out and then be prepared for mission intergration.

1.6.2.2.4 Mission Integration Testing

The modules for each mission, including pre-IOC flight test missions, will be re-qualified individually before integration. They will then be integrated into the mission configuration and tested. This will be done at the launch site in a payload/mission integration facility. This testing phase will also include the complete mission testing that will commence once the mission payload is mated with the NLS launch vehicle. This will be done at the VAB at the launch site. This testing phase will continue while the vehicle and payload are on the launch pad, up until the time of launch. The preliminary design of these testing procedures has been completed by the status subsystem for this report and can be found in Volume II, Chapter 8.

1.6.2.3 Flight Test Program

This section will outline a proposed flight test program to qualify each module and major system for flight and an IOC by the year 2000. The scheduling and cost of this program is dealt with in subsequent chapters. This section will deal with the testing purpose and methodology. The program outlined is very ambitious, but not impossible to complete.

This test program is based on a "sucessful test" philosophy. It is ambitious and its completion requires no catestrophic failures. TQM procedures and the SR&QA programs used during design and development should result in a product that can meet these ambitious qualification and testing plans.

1.6.2.3.1 Approach and Landing Tests

Multiple (6-8) approach and landing tests (ALT) will be conducted, releasing an un-piloted (at first) crew module (CM) test article from a NASA 747 or B-52 aircraft. It is recommended that two CM test articles be constructed for these tests, so different subsystems can be evaluated during separate tests. This also allows for one or two of these tests to be piloted. The location for these tests should be the proposed mission landing sites or reasonable substitutes.

The purpose of the ALTs would be to determine the low speed handling and performance characteristics of the CM and test the recovery system and the navigation and landing aids. The automated landing capability of the CM would also be verified, as well as the piloted capability in later ALTs. The possibilities for water landings and ditchings can also be evaluated during this phase.

1.6.2.3.2 Automated Docking Test

A modified lunar braking module (LBM) will be launched into LEO by an Atlas/Centaur vehicle to conduct an automated docking manuever with the Space Shuttle. The Shuttle will already be in orbit due to a previously scheduled flight whose primary mission is not the LBM docking (this will reduce cost). A simple docking ring set up in the cargo bay or on the manipulator arm would be the target for the LBM. This test will verify the hardware/software developed for the automated docking maneuvers needed in the Columbiad Program.

1.6.2.3.3 First Launch: PTLI Stage

The Primary Trans-Lunar Injection stage will be launched into LEO by an NLS vehicle. The PTLI will maintain orbit and automatically rendezvous and dock with the second launch. The PTLI will then place the payload on the lunar trajectory.

This launch will verify the spacecraft/launch vehicle interfaces and performance. It will provide for an in-flight test of the PTLI propulsion system and will verify cryogenic storability. Automated docking and trans-lunar boost ability will also be demonstrated.

1.6.2.3.4 Second Launch: Un-Piloted CM/ERM/LBM Configuration

An un-piloted CM/ERM/LBM configuration will be launched into LEO by an NLS vehicle. The stack will rendezvous and dock with the PTLI stage already in orbit from the first launch. It will travel to the moon and perform the lunar braking maneuver. The LBM will separate and the ERM will fire for placement into an Earth return trajectory. Completing this, the ERM will separate and the CM will reenter the atmosphere and land.

This launch will demonstrate the rapid turnaround and launch capabilites of Launch Complex 39, which will be used for the actual missions as well. Spacecraft/launch vehicle interfaces and performance will be tested and verified. It also allows for in-flight testing of the LBM and ERM propulsion systems. Additionally, the characteristics of the high-speed reentry of the CM will be measured and evaluated. This launch will verify all the subsystems and major systems of each module in a simulated mission environment.

At this point in time the flight test program is concluded and IOC will be achieved with the next two launches (the precursor mission). The unlikely failure of that mission would cause the testing and operational plan of the Columbiad Program to be re-evaluated.

1.6.3 Launch and Ground Support Philosophy

Once the mission payload stack has been integrated and is ready for launch it is imperative that all connected systems and modules be thoroughly tested to insure their correct operation. It will be necessary for NASA to assign a separate Status Group to control and direct the necessary testing and monitoring that occurs during this phase of a mission (up to and including launch). There are three probable types of tests that will form the basis of the launch and ground support philosophies for the Columbiad Program. These are pre-launch testing, the countdown demonstration test, and the integrated launch system test. On-the-pad testing and monitoring for an actual launch will also play a role in this philosophy. Details on these topics have been discussed in Volume II, Chapter 8 of this report.

The remainder of this section will include a discussion of managing and improving launch and launch support efficiency, along with a discussion of mission specifics that may affect these activities.

1.6.3.1 Improvement of Launch Operations Efficiency

One of the major objectives of the Columbiad Program is to achieve the program goals while minimizing costs. This is to be done through the utilization of existing technologies, quality management techniques, and the promotion of efficiency at all program levels. This efficiency is defined in terms of labor costs, labor productivity, and adherence to mission schedule.

The Columbiad Program is intended to provide a foothold for a permanent Lunar presence and the possible further exploration of space. Therefore, the true success of the program will be identified in its flexibility in being able to meet future demands. This flexibility must also be mated with reliability, adherence to scheduling, and cost accountability, as well as acceptable performance. This requires a launch operations efficiency philosophy, which is presented in the remainder of this section.

1.6.3.2 Launch Operations Management

The Columbiad Program will strive to increase efficiency at all program levels. This can be accomplished through the implementation of several common sense management techniques based on TQM principles.

A schedule that is based upon common procedures must be initially constructed. Regardless of the mission, there are certain tasks which must always be performed. There has to be an allotment of time for tasks such as routine inspections, scheduled repairs, and typical payload loading processing, as well as pre-launch activities.

Additionally, advanced computing power must be utilized to accomplish standard operations and maintenance procedures. This would also provide a database that could quicken the decision making process of various functions by providing necessary information at a moments notice.

An integrated checkout system would help alleviate unnecessary and repetitive pre-flight operations and personnel, especially those that are not critical to mission success. This is an important part of reducing extra expenditures due to over-staffing and unnecessary redundancy.

It is recommended that NASA implement operations management philosophies along the guidelines presented here, in order to streamline the management process, reduce manpower, and promote general program efficiency.

1.6.3.3 Mission Specific Effects

The efficiency of the launch operations will be influenced by the nature of each specific mission in the Columbiad Program. The best way to avod mission specific effects is to include a complete ground support plan in the planning documents of a mission. Such a plan for the support of a mission should be in place at least four years prior to the first launch. This will ensure that all of the support requirements are met . Sufficient preparation in this area will reduce the operational delay effects of specialized missions. The goal is to ensure the most efficient operation that is possible. Any changes or modifications to the ground support plan should be documented in detail. The support schedule and its addendum should be subject to a preliminary review at least one year prior to launch while a final review is made six months later. The approved flight support plan should include a detailed description of the campaign profile, vehicle development, and the standard maintenance requirements. The addendum to the support plan will in all likelihood detail mission specific challenges to the flight operations. This could include the design of flight specific software or unusual payload configurations.

1.6.4 Maintenance and CM Refurbishment

The maintenance of testing and launch facilities will be the responsibility of NASA and the centers and units directly in control of such facilities. The Columbiad Program Office will provide input and guidelines for the management of the facility maintenance. An approach for efficient facility and equipment maintenance is proposed in this section.

1.6.4.1 Efficient Maintenance Approach

Launch operations can be significantly improved with the use of a revised maintenance program which is based upon standard airline operation techniques. This reliability centered procedure works on the premise that hardware failure is usually the result of cycle use, environmental exposure, or accidents. Whatever the cause, hardware is redesigned until its performance is acceptable. This technique, in conjunction with space vehicle processing activities, can be used to improve both the reliability and maintainability of hardware at reasonable costs. Furthermore, this method allows for the analysis of failure modes. With this knowledge, schedules can be modified to include provisions for expected maintenance based on a historical data base.

Another facet of efficient maintenance handling is the procurement and inventory of spare parts. Ideally, a parts procurement program would determine a need versus current inventory status of various parts. The STS incorporates such a program in the Shuttle

Inventory Management System (SIMS) which controls the acquisition of spare parts. Spares management is usually handled by the vehicle design centers. At Kennedy Space Center (KSC),the proposed launch site of Columbiad,the upkeep of line replaceable units (LRU) is handled in the facilities' shops and labs. The Columbiad Program must incorporate a comparable program in order to follow an efficient maintenance philosophy.

1.6.4.2 Definitions

Spare parts refers to any material that is needed or will be needed to replace any assembly, subassembly, component, etc. during the operation, maintenance, repair , or overhaul of a piece of equipment.

The Spare Parts Selection List (SPSL) lists all spare parts and the price of their procurement or fabrication.

The Priced Spare Parts List (PSPL) is the final and approved version of the SPSL . It includes total quantities and firm unit and total prices.

Repair refers to the partial disassembly,modification,and test of various components or spares. It typically includes day-to-day maintenance that is performed at the test or launch site.

Overhaul will usually be performed at the manufacturing facilities of the vehicle. It involves the total disassembly and maintenance of components which have deteriorated or worn out.

Modification occurs when a component is physically altered in an effort to change its performance.

1.6.4.3 Maintenance Program Content

The development of an SPSL is the first major step of the maintenance program. It is essential that all procurements are based upon the guideline of providing required support at the lowest possible inventory level. This will minimize the potential for obsolescence that may be caused by design or engineering changes. Furthermore, the driver for determining inventory levels should be the anticipated utilization. Any shipment which surpasses this level should only be made if it is clearly in the best interest of the program.

Inventory costs must be minimized. This can be accomplished by stocking the relatively low cost items (repair/overhaul and modification kits) instead of the relatively high cost items (assemblies and modules). In addition, the economical use of repair and modification practices can also lower the the stock level. Additionally, existing assets can be drafted into service. Some of these include test components or equipment that may no longer be in use.

Launch maintenance efficiency could be markedly improved through the use of a "critical-to-launch" spare parts list. Such a list would detail the availability and quantities of launch critical replacement components during the 30-day period prior to a scheduled launch. In addition to this, a spare parts modification program (SPMP) could provide the flexibility that is required of a successful maintenance program. Such a program would assure the continued compatibility of the spares design program with the continually changing launch configuration.

One of the simplest means of improving operations efficiency is through the training and retraining of flight personnel . This can allow tasks that are handled by professionals or engineers to be done by technicians at a lower cost.

1.6.4.4 CM Refurbishment

The CM is the only module that is reusable and will be refurbished after each mission. This procedure has yet to be identified or detailed. Preliminary plans call for a similar procedure as that for the Space Transportation System (STS). It will include a complete post-flight check-out followed by repairs/refurbishment as necessary to achieve operational readiness. The CM will then undergo standard module testing prior to mission integration for the next flight. This turnaround time has yet to be estimated. Assuming a similar procedure and facilities as the STS, the time should be comparable.

1.7 Proposed Procurement Approach

NASA will procure hardware for the Columbiad Program in a manner designed to accomplish agency-wide goals. The acquisition policy that is eventually adopted should be keyed to the policy of NASA performing the SE&I in-house, as previously discussed. Recognizing that the Columbiad Program will be constrained by the availability of budget authority, the program should be based on a design to cost approach.

1.7.1 Initial Procurement

The initial procurement for a program such as Columbiad would involve conceptual definition and a preliminary design. However, much of this definition phase work has already been done by HAC and presented in this report. It would of course be advisable to have this preliminary design reviewed and revised by potential system contractors and NASA. To do this it is recommended that NASA release a single Request for Proposal (RFP) for the following module pairs: CM and ERM, Habitat and PLM, and PTLI and LBM. These pairs are grouped because of their commonality and/or design dependence. Fixed-price definition contracts would then be awarded for the work packages defined in the RFPs. Contract selections will be made by the program administrator. Management will be the responsiblilty of the Columbiad Program Office.

It is imperative that NASA carefully oversee all definition phase contractual work. This is necessary because cost must be minimized by limiting the amount of re-work done on designs already completed for this report. Additionally, the separate contracts for module pairs makes communication between companies and module pairs difficult. As long as NASA manages carefully, potential problems with the multiple module pair system can be avoided. It is desirable to have these muliple contracts in order to spread the workload and contracting profit throughout the aerospace industry.

1.7.2 Develolpment Procurement

Competition for the development phase contracts should be limited to those involved in the definition phase, unless it is in the best interest of the program to alter this approach. It is recommended that the RFPs be constructed in a similar manner as for the initial procurement, unless a change is needed and recommended. Contractor selections will be made by the program administrator. Management will be the responsibility of the Columbiad Program Office.

1.8 Proposed Resources Approach

This section describes a possible resource management approach for the Columbiad Program. It consists of descriptions of the budget process, the budget itself, facilities, and manpower.

1.8.1 Resource Management

An approach needs to be developed for the Columbiad Program which ensures that program implementation is consistent with established cost restraints. Lessons learned from other programs need to be reviewed to ensure that maximum benefit is gained from the past experiences of NASA. An approach utilizing Program Definition Reviews, such as those originally designed for the Space Station Program, is recommended.

Program Definition Reviews will be held prior to the inclusion of the Columbiad Program implementation requirements in the NASA budget request. These will insure that all elements of the Columbiad Program are well defined and understood. This approach will allow knowledgeable individuals with no vested interest in the Columbiad Program to critically review and evaluate the proposed program and plans and recommend appropriate changes before major NASA commitments.

1.8.2 Budget Process

A budget process needs to be developed for the Columbiad Program that utilizes new and innovative as well as proven cost management techniques to achieve the best program available within cost constraints. Reporting, management control and visibility, and cost assessments need to be incorporated into a comprehensive information system.

1.8.2.1 Budget Formulation

The Columbiad Program Director will establish budget guidelines including reserves for program cost growth and changes. These can be based on the cost estimates provided by this preliminary design in the next chapter. Each division of the Columbiad Program Office will develop their own program budgets and submit them to the Director for each annual NASA budget submission. Due to uncertainties concerning budget lead time, it is recommended that all parties involved in the budgeting process provide appropriate reserves for contingencies.

The budget will be evolutionary in its formulation. Initial estimates will be provided in the preliminary design, as they are in this report. These estimates will include both research and development costs as well as production costs and the program campaign. As the program matures and development contracts are awarded, the budget will be formulated on detailed engineering build-up estimates. The budget will also be modified where required by results or program independent assessments performed by the Columbiad Program Office.

1.8.2.2 Budget Allocation

Upon the approval of the budget operating plan by the Columbiad Program Director, the program managers within the Columbiad Program Office will implement the budget allocations.

1.8.2.3 Budget Statusing

A state-of-the-art resources information system needs to be developed by the Columbiad Program Office to provide current status of program costs to managers at all program levels. This system (to be developed or procured) must emphasize timeliness, completeness, and accuracy. It must also be structured to emphasize abnormalities that would require special management attention. Periodic Program Review Meetings, internal to NASA, should be held at a schedule to be determined by the Director. At these meetings cost variances and total projected costs would be reported by the program managers to the Columbiad Program Director.

1.8.3 Budget and Cost Approach

In general, program funding profile and schedule need to follow a disciplined technical and management approach to insure that the transition from development to production is smooth and to assure the proper overlap of development and early production funding. Figure 1-4 illustrates the cost curve which is desired.

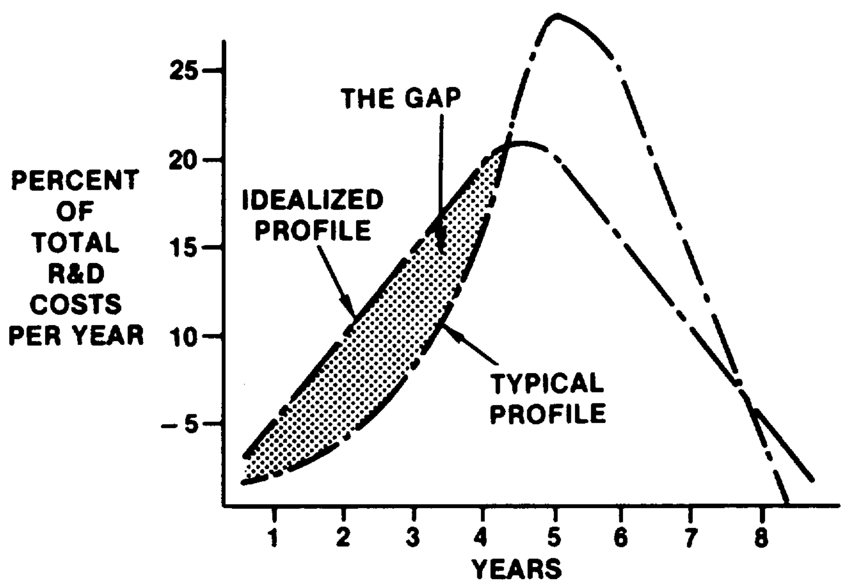

Figure 1-4
General Desired Cost Curve with Undesired Design Gap

This figure shows the desired cost profile and the typical cost profile for a general case, not specific to the Columbiad Program. Specific profiles have not been developed within this preliminary design, although direct costs are discussed in Chapter Two of this volume. It is desired that the design gap encompass as little area as possible so that spending can be smoothly distributed throughout the program. This gap can be lessened by making performance and cost trade-offs before commiting to a design approach, and making sure that a production plan and production cost estimates are reasonable.

1.8.4 Facilities

The Columbiad Program may require some funding for the design and construction of new facilities. This may be especially true for some of the subsystem integration needs of the program as well as the module integration needs. Maximum use will be made of existing or modified existing facilities. The following activities and functions will require specialized facilities: technology development, subsystem development, manufacturing, testing, design verification, training, checkout, servicing, integration, launch, and mission support. The exact level of modification needed to existing facilities to meet these needs has not been determined for this preliminary design. It is anticipated that the number of new facilities will not be great, due to the extensive use of existing technology within the design of the Columbiad Program.

A long-range facility plan would have to be designed by the Columbiad Program Office and reviewed by the Director as soon as possible following the final preliminary design. This plan should be reviewed regularly and modified annually as necessary.

1.8.5 Manpower and Training

The resource of manpower will be an integral part of the Columbiad Program. Civil service, military, and contracted personnel will be represented in all phases of the program. There will be a need for a division of the Columbiad Program Office to handle employment and training of these employees for the specific aspects of the program that they are responsible for. A specific plan for this has not been determined for this preliminary design. Contractors will be responsible for specific training of their employees for work they have been awarded contracts to complete.

The training of the crew and operational support teams for launch will be the responsibility of NASA and its various components. Astronaut selection will be structured around the current STS system. Training will involve general space training as with current astronauts as well as mission-specific training using simulators to be developed by the module contractors. An exact training plan has not been developed at this time. Launch support team training will be a modification of existing programs, since the planned launch facilities and activities closely follow the existing STS program.

2 Cost Estimation and Analysis

Cost is an important design variable for the Columbiad Program, behind only crew safety in design criteria. This drives the design philosophy which dictates that component and system commonality, and low-risk technology are to be used in the design whenever possible to cut down on development and integration costs as well as technological uncertainty factors. The life-cycle costing of this project is broken down into several levels. In the technical and engineering level, cost is divided into six areas:

- Research, Development, Testing and Evaluation (RTD&E)
- Production
- Transportation
- Mission Support and Operations
- Maintenance and Refurbishment
- Training of Astronauts

In the administration and systems integration level, additional costs are considered in two areas:

- General and Administrative
- Profit

Cost is broken down into three general types: nonrecurring, recurring, and operations. Nonrecurring costs include the RTD&E, the training of astronauts and other personnel, and the general accounting and profit associated with such costs. Recurring costs refer to the production of mission stages and launch vehicles. Here, learning curve effects are considered where appropriate. Operations costs are those related with mission support, crew module maintenance and refurbishment, and administrative factors associated with such support.

2.1 Cost Descriptions

This analysis assumes a program horizon of eight years RDT&E and production time, and five years of campaign in which five precursors and fifteen piloted flights are operated. All costs are estimated in 1992 dollars. Cost estimation is calculated from both analogous and bottom-up approach, where analyses down to the subsystem level is done to better define the cost estimates. The decision to use analogous and bottom-up estimation seems more reasonable than using parametric estimation since most components are chosen for their high technological certainty and availability, and prices can be readily obtained from suppliers or technical papers. This section will define the costs taken into account in this analysis, and the next section will describe the actual process by which the computation is made, with the actual numbers assumed in this process. Then, cost estimation results for each module is summarized. Detailed cost assumptions for each subsystem is given in the Appendix.

2.1.1 Research, Development, Testing and Evaluation

Research, Development, Testing, and Evaluation (RDT&E) Cost encompasses all costs prior to the actual production. This includes engineering design, construction of operational prototypes for test integration, and improvement on the design. Such considerations as government test and contractor support must be taken into account. As a first-cut analysis to be used as a sanity check, each stage or module is estimate to cost between 0.5 billion and 2 billion depending on the mass, size, and complexity of the stage. The "stages" are divided such that every phase of the mission is considered separately:

- Crew Module
- Earth Return Module (ERM)
- Lunar Braking Module (LBM)
- Primary Trans-Lunar Injection (PTLI) Stage
- Surface Payloads
- Payload Landing Module (PLM)
- Launch Vehicle

Within each module, subsystems costs are compiled.

Technology readiness of every subsystem component is taken into account in the RDT&E of each module. Five levels of Technological Readiness are defined in Table 2-1.

Table 2-1: Technology Readiness Definitions

Technology Level	Definition
1	Out on the market and available from supplier
2	Tested and ready for production
3	Prototype has been developed for testing
4	Technology available but RDT&E needed
5	Hunsaker Aerospace Corporation (HAC) design or proposal

Most of the hardware chosen for Project Columbiad are Level 1 and cost is calculated according to the prices quoted from suppliers. For these products, RTD&E cost is assumed to be covered by the price, and only handling or storage facility costs are additionally considered. Technology levels of 3 to 5 obviously require more than the cost of production. These are taken into account through RDT&E estimations.

RDT&E costs are categorically non-recurring. The only phase of the mission which required the use of RDT&E cost more than once is the Surface Payloads. The same surface equipment is not sent up with each precursor mission, so additional RDT&E must be considered for these.

2.1.2 Production

Production cost (Prod) is the cost associated with the manufacturing and delivery of the system and subsystems in the quantity specified. This cost is highly dependent on the quantity which is produced since a "Learning Curve Factor" is applied to account for productivity improvements. The total production cost for *N* units is modelled as:

$$Production\ Cost = TFU \times L$$
$$L = N^B$$
$$B = 1 - \frac{\ln(100\% / S)}{\ln 2}$$

where

TFU:	the theoretical first unit cost
L:	the learning curve factor
S:	the learning curve slope in percent
B:	factor set by *S*

Because of the complexity and uncertainty of applying such a systems cost estimation to each individual subsystem component, this approach is only applied to the module level. All costs associated with a particular module are calculated and compiled before the Learning Curve Factor is used for the entire system in calculating the campaign cost.

2.1.3 Transportation

Transportation cost (Trans) is the cost to transport the entire precursor and piloted mission from the launch pad to Low Earth Orbit (LEO). This is divided up into three sections: the launch vehicle production, the launch operations and the support cost. The transportation cost will not include any operations and support costs associated with transit from LEO to the Moon, the lunar stay, nor the transit back to the Earth. These costs are included in the mission support and operations described below. Fuel cost will not be considered in great detail here nor in the mission after launch since it is not expected to influence the cost of the transportation by a great amount.

2.1.4 Mission Support and Operations

Mission Support and Operations (S&O) include the operations and ground support for the entire one-month mission. This is determined mainly by the amount of support which the status group considers essential for monitoring the mission.

2.1.5 Maintenance and Refurbishment

Maintenance and Refurbishment (M&R) is a very important consideration since the crew capsule is designed to be reusable. This cost must account for the maintenance and possible replacement of certain subsystems in the capsule.

2.1.6 General and Administrative

General and Administrative (G&A) Costs are those administration and management costs associated with the system integration, quality, and compatibility. The factor which is used in this analysis is taken to be 12% of the entire RTD&E and production costs.

2.1.7 Profit

Commercial contractors will be involved in this program, and profit is a basic consideration. This is usually taken to be anywhere from 10-15% of the entire cost of their project, including RTD&E, production, and G&A where applicable. In this analysis, a 15% factor will be used.

2.1.8 Training and Simulation

The lunanauts will require additional simulation equipment than those previously provided for space missions. A one-time cost of the modelling and development of a simulator as well as the general training of the astronauts must be calculated.

2.2 Cost Assumptions and Computation

Several levels of nesting is required for the cost analysis. Figure 2-1 shows the cost analysis process which went into calculating the cost of each module.

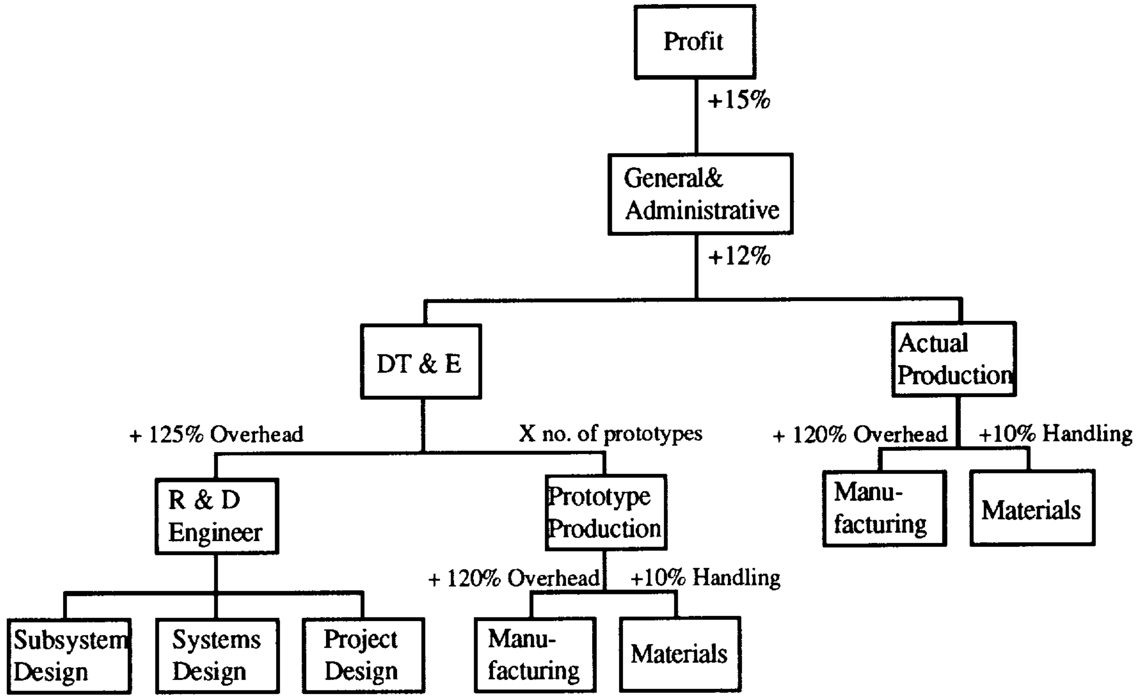

Figure 2-1
General Cost Breakdown for One Module

The two most important components of cost are the RDT&E and the actual production cost of the module. The actual production cost is a compilation of all subsystems costs in the module. This section describes the cost assumptions and computations on a module system level. The subsystems assumptions and costs are presented in Appendix 4-1.

Engineering and prototype production are included in the Development, Testing, and Evaluation (DT&E) costs. The Research and Design (R&D) Engineer is then divided up into the system, project, and subsystem sections. The subsystem section will be further divided into the applicable groups for each module. Per engineer cost per year are assumed to be $60 thousand dollars to take into account both the engineer's salary and benefits. An overhead of 125% is added for additional support costs like administration and computer support. According to resources the number of engineers who are involved in a particular subsystem in a module falls anywhere between 12 and 30 depending on the complexity and involvement of the subsystem in that module.

Under prototype production falls both the cost of materials and of manufacturing. Here, distinction is made between level 1 components and components with other Technological Uncertainty factors. Level 1 hardware requires only the addition of 10% its original price to account for the handling and storage requirements. For this report, the assumption has been made that Level 2 components will also fall under this category, although a theoretical first unit cost will take the place of an actual quoted price. For the Level 3 to 5 components, not only is the expected materials cost increased by 10% for the proper handling and storage, but a manufacturing cost is also taken into account. For manufacturing a wage of $20 per hour per worker is assumed. Then, on top of the worker production costs, a 120% overhead is included for the general facility costs.

A general and administrative cost of 12% is a reasonable addition for most management and administrative factors. Finally, assuming that each module is distributed to a commercial contractor, a fractional factor of the entire cost is included to account for profit.

The total cost for the first launch is calculated from the combination of all RTD&E and production costs for each module. The breakdown is presented in Figure 2-2.

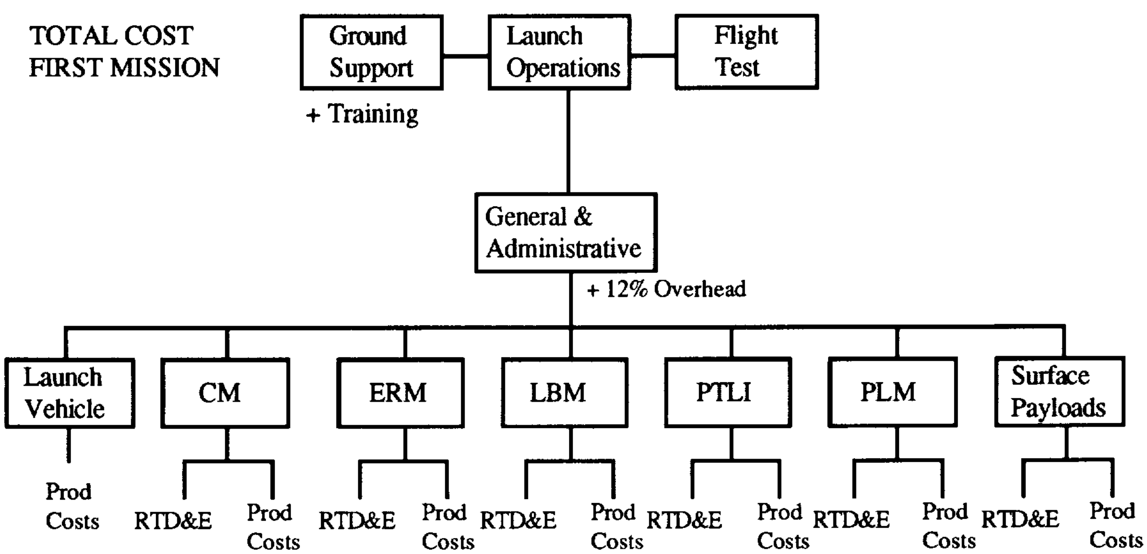

Figure 2-2
Total Cost Breakdown for First Mission

After the completion of each module, management costs of 12% is again added on top of total module costs to account for the integration of all stages, including flight tests and ground tests. These costs could be thought of as the top level management costs associated with the integration authority NASA. Finally, launch operations and ground support is added.

The first mission includes both a precursor mission and a piloted mission, but additional missions use only a piloted mission. Right now, a precursor mission is scheduled for each year with three piloted missions a year. The entire Columbiad campaign is expected to continue for five years. Figure 2-3 demonstrate the campaign cost breakdown for the first two years. The third to fifth years follow an identical outline.

Total campaign cost is calculated from this. RDT&E cost is a one-time cost for each stage except for the surface payloads, where five distinct developments are assumed for the five different precursor payloads to be placed on the moon. The production cost is factored with a learning curve, where a 95% learning curve (refer to equations in section 2.1.2 for L =95%) is used for those modules which are produced under a total of ten times. For more than ten units of production, a 90% learning curve is used. The total program cost is thus calculated to be $46.4 Billion without any margins applied.

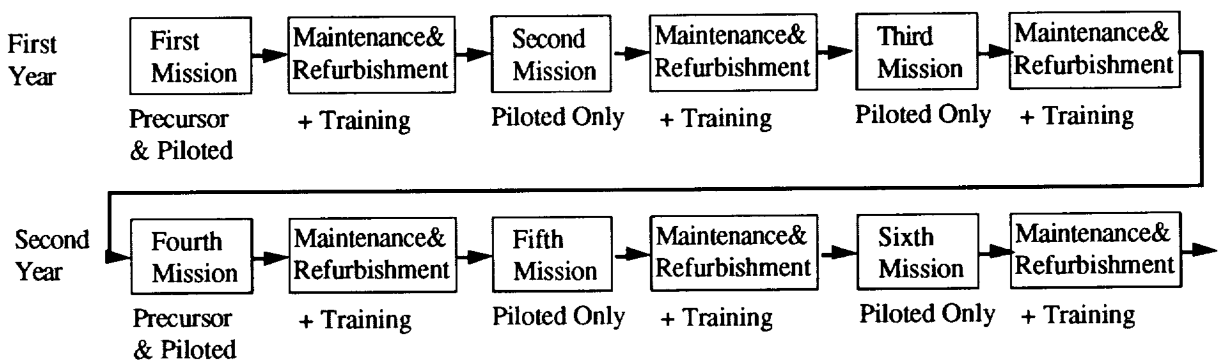

Figure 2-3
Campaign Cost Breakdown

2.2.1 Crew Module

The crew module involves the cooperation between crew systems, structures and thermal protection, power and thermal control, and the avionics groups. Several assumptions about the involvements of engineers and manufacturing workers are made. Figure 2-4 gives a quick summary of the assumptions for the RDT&E costs.

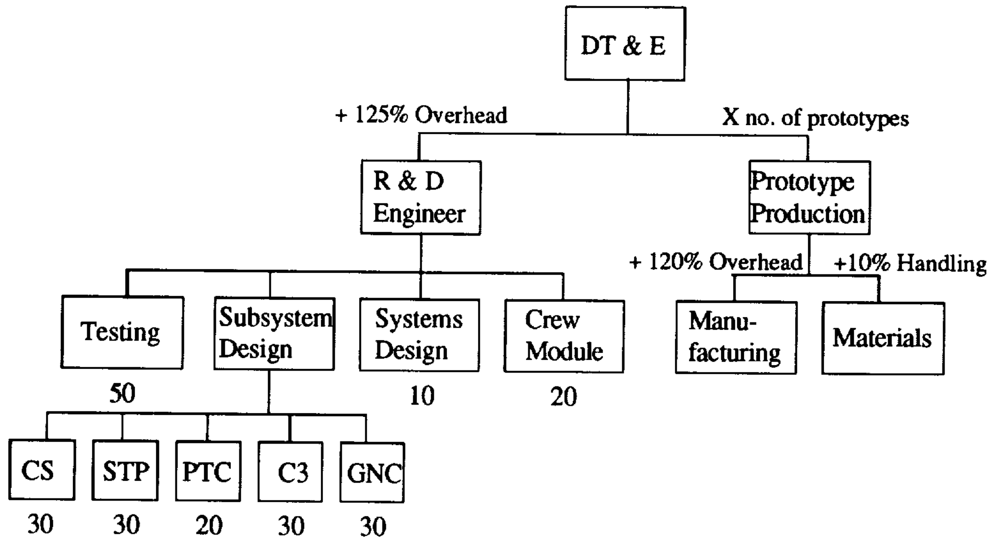

Figure 2-4
Crew Module Cost Breakdown

The numbers which fall below each of the engineering categories indicate the number of engineers which are predicted to be necessary for each subsystem. Each of these engineers are expected to contribute an average of five years to see the project develop from the design phase through to the end of the evaluations except for the additional engineers which are needed for the conduction of tests. These engineers are expected to contribute only around three years to the development and testing portion of the RDT&E.

Table 2-2 on the next page shows the major costs associated with the Crew Module. All costs are in millions and include the handling and overhead factors. The RTD&E cost includes the manufacturing of three prototypes, and the salary for the 220 engineers who are expected to design and evaluate the module. The breakdown of production costs include a separation between the set manufacture's prices and the production costs of a theoretical first unit. The ejection seat, reaction control propulsion system, and portions of the equipment required by the crew have already been developed and can be obtained from suppliers. The other components all need to be developed further and, therefore, theoretical unit prices are much higher. Total production cost for the entire module is $105 Million, and the total expected RDT&E cost is $685 Million.

Table 2-2: CM Costs

<table>
<tr><th>Subsystem Component</th><th>Set Prices</th><th>Theoretical First Unit Price</th><td rowspan="3">Production Cost
105.00</td></tr>
<tr><td>Ejection Seat</td><td>3.30</td><td></td></tr>
<tr><td>R4-D engine</td><td>1.76</td><td></td></tr>
<tr><td>Crew Systems</td><td>6.58</td><td>35.49</td><td rowspan="4">RTD&E Cost
426.73</td></tr>
<tr><td>Structures</td><td></td><td>18.70</td></tr>
<tr><td>Heat Shield</td><td></td><td>3.74</td></tr>
<tr><td>Wing& Deploy</td><td></td><td>3.30</td></tr>
<tr><td>Landing Gear</td><td></td><td>1.10</td><td rowspan="3">RTD&E with G&A
595.53</td></tr>
<tr><td>Paraglide</td><td></td><td>1.32</td></tr>
<tr><td>GNC</td><td></td><td>8.86</td></tr>
<tr><td>C3</td><td></td><td>11.94</td><td rowspan="4">Total RTD&E Cost with Profit
684.86</td></tr>
<tr><td>Batteries</td><td></td><td>4.51</td></tr>
<tr><td>Software</td><td></td><td>4.41</td></tr>
<tr><td>Prod Cost</td><td>11.64</td><td>93.36</td></tr>
</table>

2.2.2 *Earth Return Module*

The Earth Return Module involves the cooperation between the propulsion, structures, power and thermal protection, and the avionics groups. Only 12 engineers are needed for one year in the propulsion subsystem group since the propulsion system has basically been chosen, and these engineers just need to help incorporate the system into the module. More engineers are required for the development of structures and power since both subsystems involve relatively new technology. Figure 2-5 summarizes the assumptions for RDT&E.

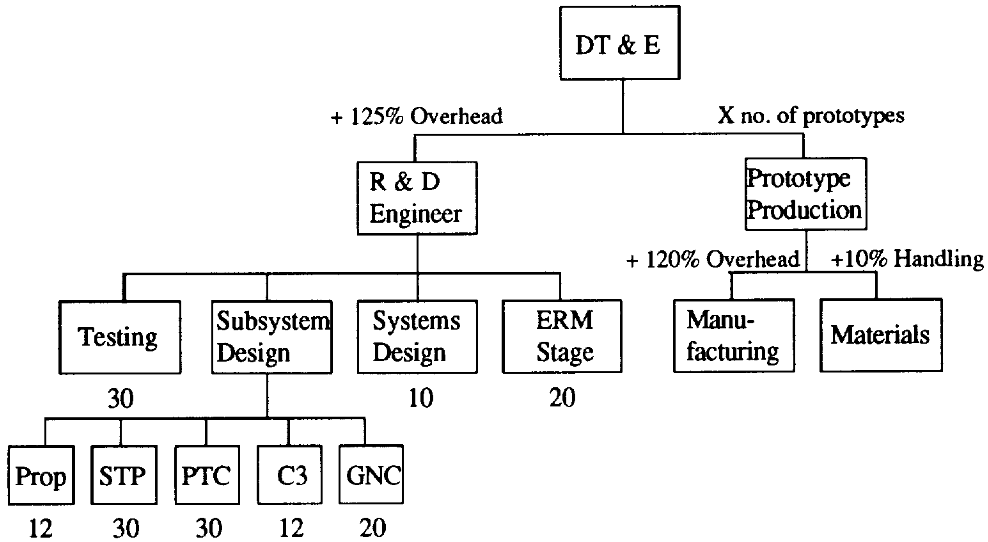

Figure 2-5
Earth Return Module Cost Breakdown

Table 2-3 shows the RDT&E and production costs for the ERM. Only the two propulsion engines, RL10A-4 and R4-D, are available from suppliers. Everything else requires further development and testing.

Table 2-3: ERM Costs

<table>
<tr><th>Subsystem Component</th><th>Set Prices</th><th>Theoretical First Unit Price</th><th rowspan="2">Production Cost

83.94</th></tr>
<tr><td>Casing</td><td></td><td>19.25</td></tr>
<tr><td>Truss</td><td></td><td>4.46</td><td rowspan="3">RTD&E Cost

313.36</td></tr>
<tr><td>Tanks</td><td></td><td>13.42</td></tr>
<tr><td>RL10A-4 (3)</td><td>6.60</td><td></td></tr>
<tr><td>R4-D (16)</td><td>3.52</td><td></td><td rowspan="3">RTD&E with G&A

444.98</td></tr>
<tr><td>GNC</td><td></td><td>7.24</td></tr>
<tr><td>C3</td><td></td><td>5.32</td></tr>
<tr><td>Power</td><td></td><td>18.86</td><td rowspan="3">Total RTD&E Cost with Profit

511.73</td></tr>
<tr><td>Miscellaneous</td><td></td><td>5.28</td></tr>
<tr><td>Prod Cost</td><td>10.12</td><td>73.82</td></tr>
</table>

2.2.3 Lunar Braking Module

The Lunar Braking Module also involve the same subsystem groups: propulsion, structures, power and thermal protection, and avionics.

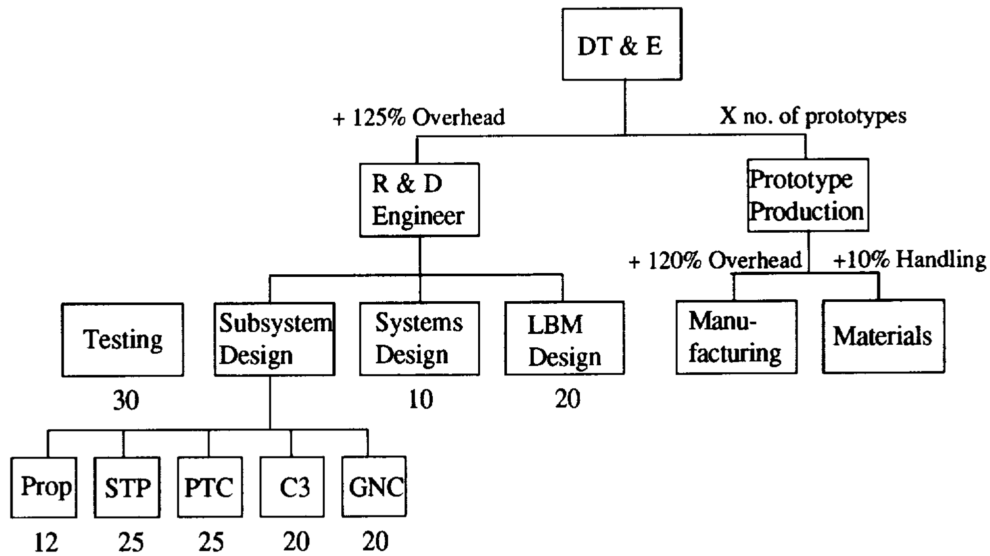

Figure 2-6
Lunar Braking Module Cost Breakdown

Table 2-4: LBM Costs

Subsystem Component	**Set Prices**	**Theoretical First Unit Price**	**Production Cost**
Casing		19.58	**73.11**
Truss		4.46	**RTD&E Cost**
Tanks		13.75	
RL10A-4 (3)	6.60		**296.84**
R4-D (0)	0.00		**RTD&E with G&A**
GNC		6.72	**414.34**
C3		4.40	
Power		13.20	**Total RTD&E Cost with Profit**
Miscellaneous		4.40	
Prod Cost	**6.60**	**66.51**	**476.49**

2.2.4 Primary Trans-Lunar Injection Stage

The propulsion, structures, power and thermal protection, and avionics groups are also involved in this propulsion stage.

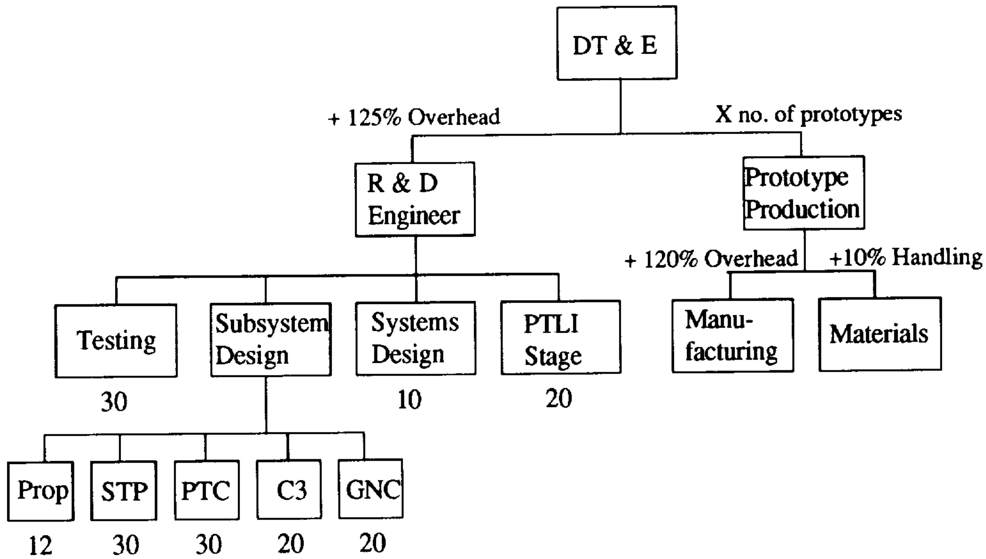

Figure 2-7
Primary Trans-Lunar Injection Stage Cost Breakdown

Table 2-5: PTLI Costs

Subsystem Component	**Set Prices**	**Theoretical First Unit Price**
Casing		20.27
Truss		5.06
Tanks		20.90
RL10A-4 (5)	11.00	
R4-D (16)	3.52	
GNC		9.17
C3		6.91
Power		18.86
Miscellaneous		3.96
Prod Cost	**14.52**	**85.13**

Production Cost	**99.65**
RTD&E Cost	**357.11**
RTD&E with G&A	**511.58**
Total RTD&E Cost with Profit	**588.31**

2.2.5 Surface Payloads

Surface Payloads are designed by the crew systems, structures, power and thermal protection, guidance, and communications groups. Since lunar modules have never been seriously developed and tested, many engineers are anticipated to participate in the design.

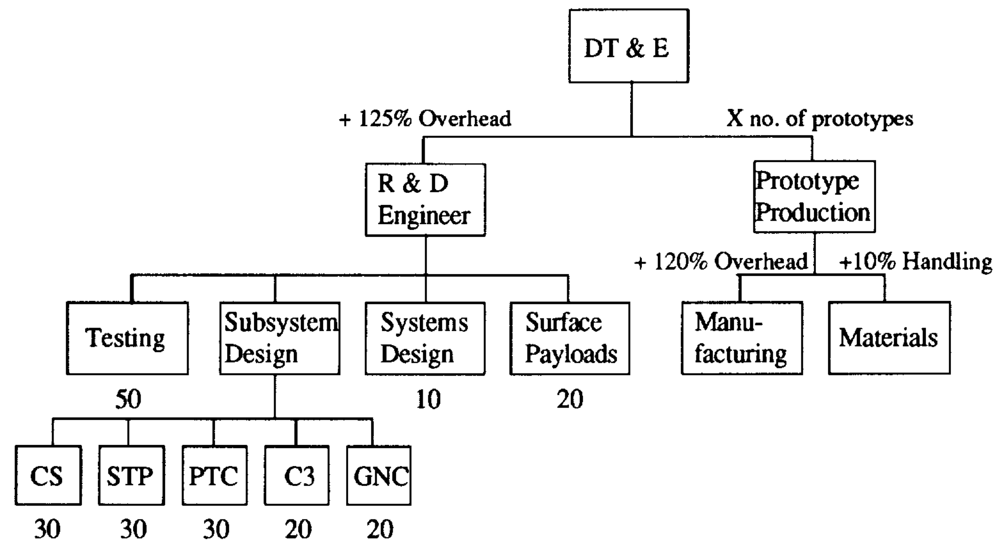

Figure 2-8
Surface Payloads Cost Breakdown

This fact also carries over to the development and manufacturing costs.

Table 2-6: Surface Payloads Costs

<table>
<tr><th>Subsystem Component</th><th>Set Prices</th><th>Theoretical First Unit Price</th><th rowspan="2">Production Cost
318.71</th></tr>
<tr><td>BioCan</td><td></td><td>46.20</td></tr>
<tr><td>Power Bay</td><td></td><td>9.30</td><td rowspan="3">RTD&E Cost
893.41</td></tr>
<tr><td>Rover</td><td></td><td>14.30</td></tr>
<tr><td>Bagger</td><td></td><td>14.30</td></tr>
<tr><td>Crew Systems</td><td>18.49</td><td>73.27</td><td rowspan="2">RTD&E with G&A
1355.50</td></tr>
<tr><td>GNC</td><td></td><td>10.88</td></tr>
<tr><td>C3</td><td></td><td>12.02</td><td rowspan="3">Total RTD&E Cost with Profit
1558.82</td></tr>
<tr><td>Power</td><td></td><td>118.10</td></tr>
<tr><td>Prod Cost</td><td>18.49</td><td>298.37</td></tr>
</table>

2.2.6. Payload Landing Module

The Payload Landing Module will again involve the participation of the propulsion, structures, power, and avionics groups. From these assumptions, the total PLM production cost is calculated to be $67 Million, and total RTD&E cost is $446 Million.

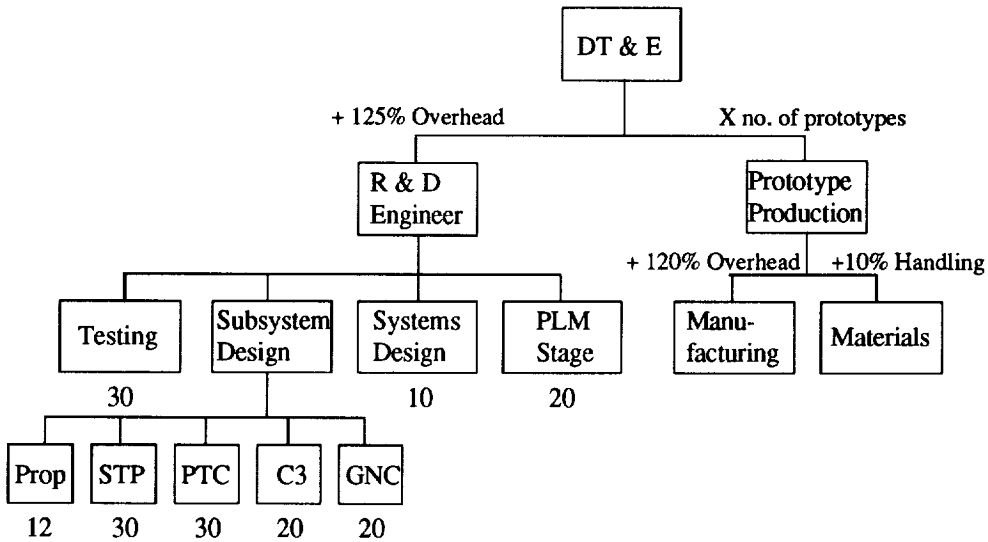

Figure 2-9
Payload Landing Module Cost Breakdown

Table 2-7: PLM Costs

Subsystem Component	**Set Prices**	**Theoretical First Unit Price**	**Production Cost 67.32**
Structure		20.90	
Landing Gear		5.94	**RTD&E Cost 279.19**
RL10A-4 (3)	6.60		
R4-D (16)	3.52		**RTD&E with G&A 388.09**
C3		6.60	
Power		19.80	**Total RTD&E Cost with Profit 446.31**
Miscellaneous		3.96	
Prod Cost	**10.12**	**57.20**	

2.2.7 Launch Vehicle

The National Launch System is expected to be developed for other missions, and therefore the Columbiad Program does not expect to use its funds to support the RDT&E costs for such a launch system. A first order cost estimate for the production cost of the NLS is calculated with the following numbers.

Table 2-8: Launch Vehicle Costs

Component Name	TFU Cost	Number Per Launch	Total Cost of Component
RSRM	38	4	152
STME	60	4	240
NLS Core	54	1	54
Nose Cone	5	1	5
Payload Adapter	5	1	5
Total for Precursor			**456**
Total for Piloted			**451**

The Redesigned Solid Rocket Motor (RSRM) is currently used for the Space Shuttle and has been priced at $38 Million. The Space Transportation Main Engine (STME) is an engine which is derived from the Space Shuttle and is currently being developed by Rocketdyne, Pratt and Whitney, and Aerojet. The production cost is expected to be slightly higher than the $45 Million dollars for the Space Shuttle engine. The core estimation is also multiplied by a factor, resulting in $54 Million. Finally, the nose cone and payload adapter is estimated to cost $5 Million each. The nose cone is only needed for the precursor mission to cover the tip-over propulsion system. This gives a launch vehicle cost of $456 Million for one launch of the precursor and $451 Million for one launch of the piloted vehicle. For simplicity, an average of these two numbers is used in the calculation of the entire mission.

2.2.8 Total Cost of First Mission

The total cost for the first mission is calculated to be $12.8 Billion. This includes the RDT&E cost and production cost of both the precursor mission and the piloted mission, launch vehicle production and operations cost, status monitoring, astronaut training and simulator development, flight tests, and general and accounting costs. Table 2-9 summarizes each.

Table 2-9: Total First Mission Cost

Piloted Mission RDT&E and Prod Costs	2623.09
Precursor RDT&E and Production Costs	2563.91
Launch Vehicle Prod Costs (4 launches)	1814.00
Launch Operations Cost (4 launches)	100.00
Training and Simulator Development	900.00
Status Monitoring for One Month	23.60
Flight Tests	4168.70
General and Accounting	622.44
Total Cost of First Flight	**12815.74**

Launch Operations Cost is assumed to be $25 Million per launch and the astronaut training and simulator developments is assumed to be $900 Million. Status monitoring costs assume that one-tenths of the activities are EVA and nine-tenths are IVA. EVA unit cost is assumed to be $84,237/hour and IVA is $29,483/hour. Both of the S&O costs are quite low compared with recent estimations because Columbiad's goal of decreasing the number of people in status monitoring was taken into account.

The total flight tests cost is calculated to be $4.2 Billion. This follows the flight test philosophy of Chapter 1 where the total modules and factors are considered: 2 CMs, 2 NLSs, 2 modified LBMs, 1 ERM, 1 Atlas, and 1 PTLI. It does not include any shuttle operations costs.

Finally, taking into account learning curve factors as well as maintenance and refurbishment costs, the following first year cost and the cost for the five year campaign is computed. Again, all costs in Table 2-10 are in $Millions.

Table 2-10: Total Campaign Cost

	Total First Year Campaign Cost	Total Five Year Campaign Cost
Piloted Flights	3076.35	5102.99
Precursor Flights	2563.91	10720.65
Launch Vehicle (8 launches)	3628.00	18140.00
Launch Operations	200.00	1000.00
Training and Simulator	1350.00	4500.00
Status Monitoring	94.40	472.00
Flight Tests	4168.70	4168.70
Total General & Accounting	709.25	1498.31
Maintenance /Refurbishment	150.00	750.00
Total Cost	**15940.61**	**46352.65**

3 Schedule

The schedule of Project Columbiad is very ambitious. The driving factor is the goal to return humans to the Moon by the year 2000. In order to achieve this, the United States must dedicate the necessary resources for the development program during the next eight years. High levels of industry and government involvment in the project are necessary to its success and timely accomplishment.

In the interest of low cost and the shortest possible development time, all designs have utilized as many existing and proven components as possible. This has helped to keep component and subsystem development time to a minimum, leaving more time for developing the modules as a whole. For instance, the propulsion systems on all of the propulsive modules use technology that is very well established. The RL-10A engines first fired in 1959, and cryogenic storage techniques have been used in many previous rocket programs.

3.1 Program Schedule

The Columbiad Program spans over a decade of activity. The development of the propulsive modules, the Crew Module, and the BioCan all occur in the first eight years. The five-year campaign of lunar surface activity begins in 2000. During this time, development of additional laboratory or scientific hardware for lunar exploration may occur. The schedules presented here do not encompass the development time for this additional hardware.

A large part of the concept development for Project Columbiad is outlined in this report. In addition, preliminary designs of each module and subsystem are presented. As soon as funding and approval of the design come through, the detailed design process can begin. This design process will continue until mid 1995. The development, testing, and evaluation process will take approximately three years, from 1995 until 1998. The production of the final components is scheduled for 1997 until 2000. The first precursor launch is scheduled for June, 2000, and the first staffed mission is set for July, 2000. The five year campaign continues until 2005. See Figure 3-1.

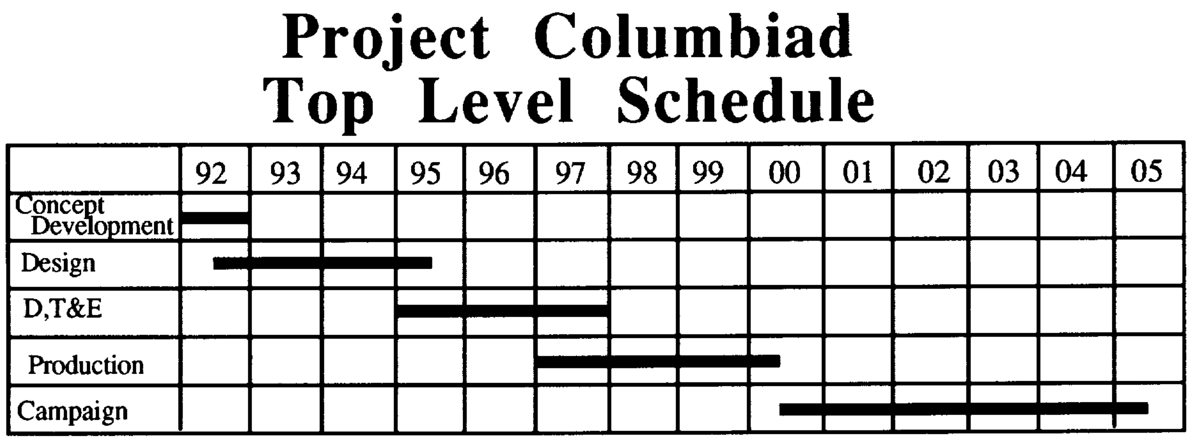

Figure 3-1
Top Level Schedule

3.1.1 Program Hardware Development

The hardware development schedule for Project Columbiad is shown in Figure 3-2. The hardware development program starts in 1992 with the detailed design programs of the Crew Module and the BioCan. The Crew Module and the BioCan will take the longest to develop, since they are the most complex modules. The design programs for the propulsive modules start in 1994. By delaying the start of the program, the cost of the development will be kept as low as possible.

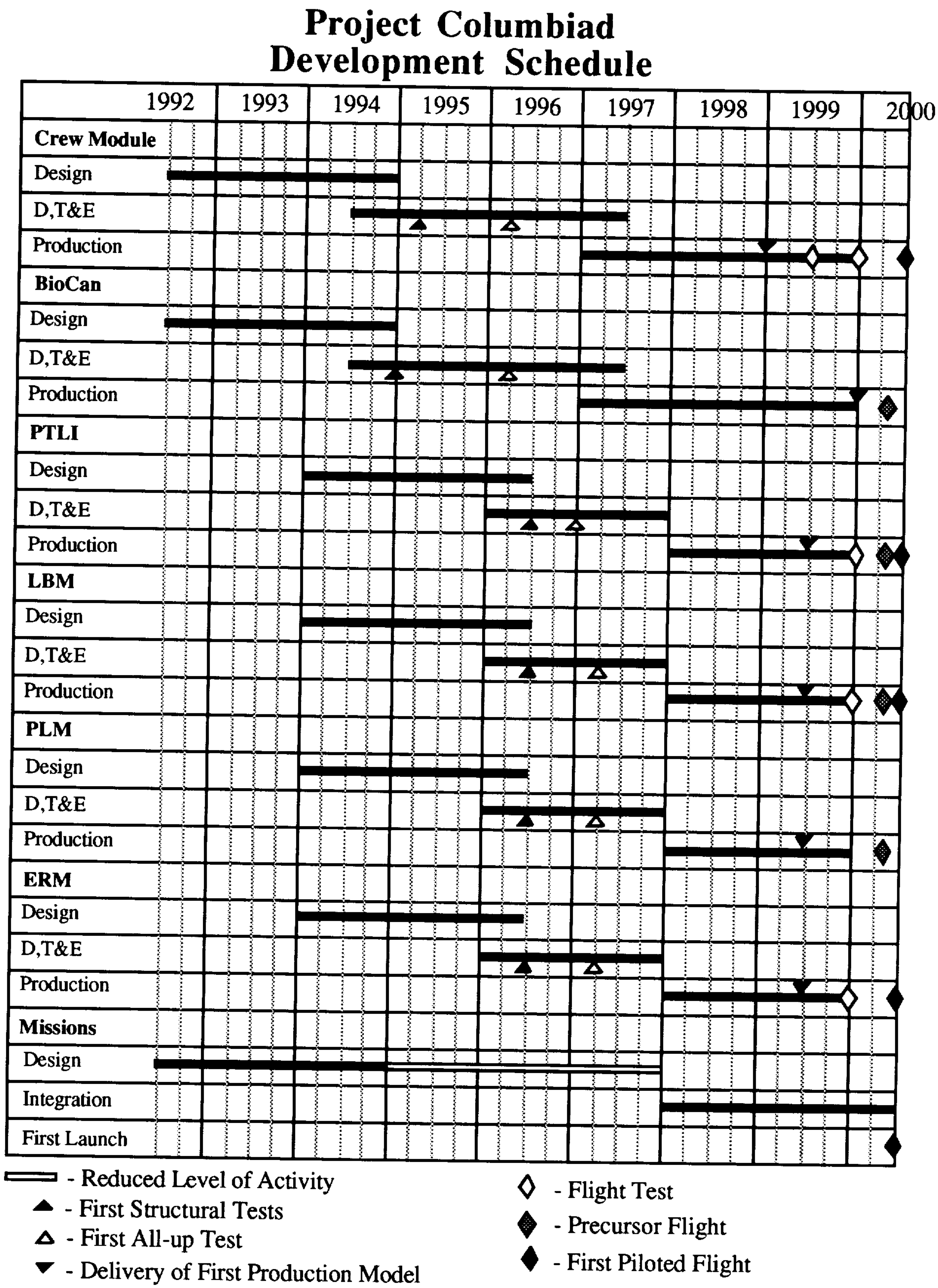

Figure 3-2

Hardware Development Schedule

3.1.2 Flight Test Schedule

The flight test program for Project Columbiad is designed to test all of the essential hardware before they fly with a crew. Some of the hardware that does not carry humans (such as the PLM) is not flight tested before the initial mission. The first flight tests of the program are designed to test the re-entry and landing capability of the Crew Module. These will occur in 1999. In the latter half of 1999, there will be an orbital test of automated docking techniques and technologies. In January, 2000, a full unpiloted test of the piloted mission is scheduled. There will be two launches of the NLS, within hours of each other. In the first will be the PTLI stage. In the second will be the LBM/ERM/CM stack. They will dock in LEO, and then travel to the moon. Instead of landing on the moon, the vehicle will circle back and the CM will reenter the Earth's atmosphere and land by automatic control The next test will be the precursor mission. This will test the PTLI, LBM, and PLM stages. At this point all of the propulsive stages will have been flight tested twice except the ERM and PLM. The similarity of the propulsive stages, particularly the LBM, ERM, and PLM, jusitifies declaring operational capability after only one test each of the PLM and ERM. See Figure 3-3 for the detailed schedule of the flight test program.

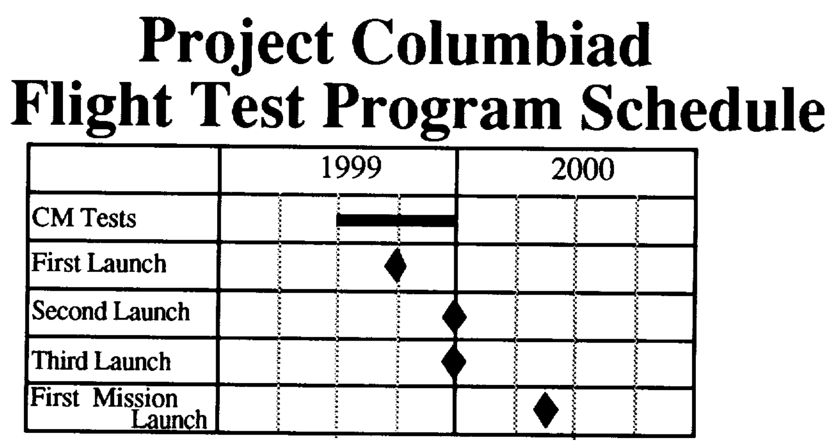

Figure 3-3
Flight Test Program Schedule

3.1.3 Campaign Schedule

The mission campaign of Project Columbiad is a five-year campaign of lunar exploration and scientific experimentation. Each piloted mission to the lunar surface will last one month, during which four crew members will live on the lunar surface. The precursor missions to the lunar surface are for the purpose of depositing larger payloads than the ERM and the CM have the capacity to carry. The goal for Project Columbiad is to accomplish three piloted and one precursor mission per year. It is assumed that the launch pads from which the NLS will launch have a launch turn-around time of one month. There will be two launch pads from which to launch the NLS to accomodate the two separate launches for each mission. Ideally, these launches will occur within hours of each other. In case of a problem with one of the launches, the PTLI stage has the capability of remaining in LEO for up to forty days before boil-off of the cryogenic fuel leaves insufficient fuel to accomplish the mission. However, since some of the missions occur within a month of each other, the schedule may slip due to failed previous launches. See Chapter 4 for a more detailed discussion of the campaign and a preliminary schedule.

3.2 Module Development Schedules

The development schedules for each module include the design, testing, and integration time for each major subsystem. In addition, integration and testing time for the entire module is included. The subsystem integration time is kept to a minimum due to the advanced level of technology available in most subsystem areas. This keeps the module development times and costs to a minimum.

3.2.1 CM Development Schedule

The biconic design of the Crew Module and the use of a paraglider to slow and control the descent are two of the least developed of all the components in the project. For this reason, the development and testing program of this module is critical to the program schedule. The detailed design of the CM will begin in mid-1992. The subsystem requiring the most development and testing time is the structure. The other subsystems are based on expertise gained in the Apollo, Skylab, and Shuttle programs. Testing of the structure will begin in 1995 and continue until 1997, when production of the final design will begin. Final qualifying flight tests will occur in 1999 and early 2000. See Figure 3-4 for the detailed development schedule.

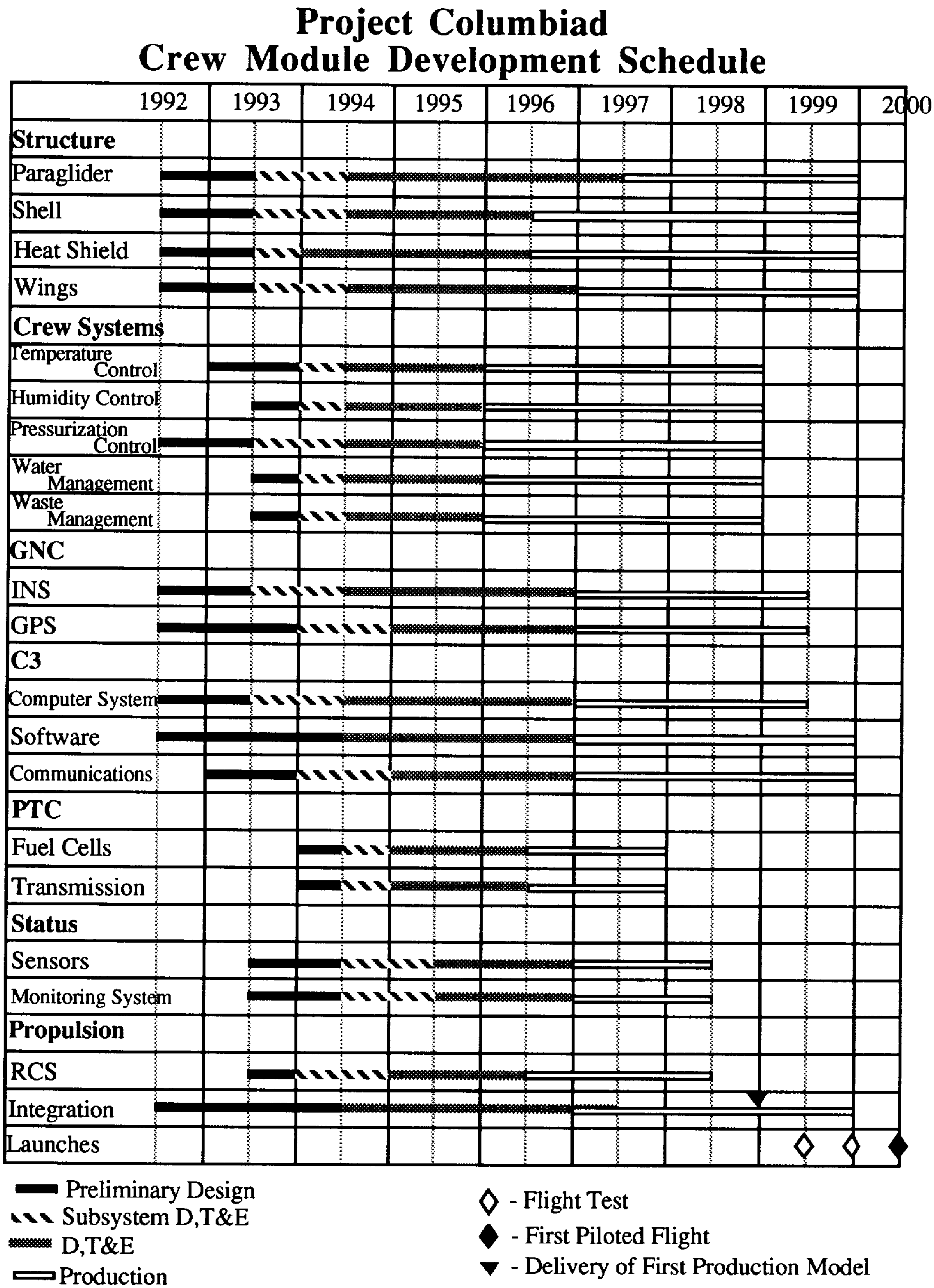

Figure 3-4

Crew Module Development Schedule

3.2.2 BioCan Development Schedule

The size and the complexity of the BioCan necessitate a longer development time for this module than for the propulsive modules. There are several complicated systems in the BioCan. The communications system, the structure, the computer system with its software, and the landing system will take the longest development time. The life support systems are all based on SpaceLab and Space Shuttle technology, so that, while complicated, the systems are well-developed. Design of the various systems will continue until about 1994. Subsystem testing and integration testing will continue through 1996, when production will begin. The BioCan will be ready for delivery to the Moon in 2000. The schedule of the development is outlined in Figure 3-5.

Project Columbiad
BioCan Schedule

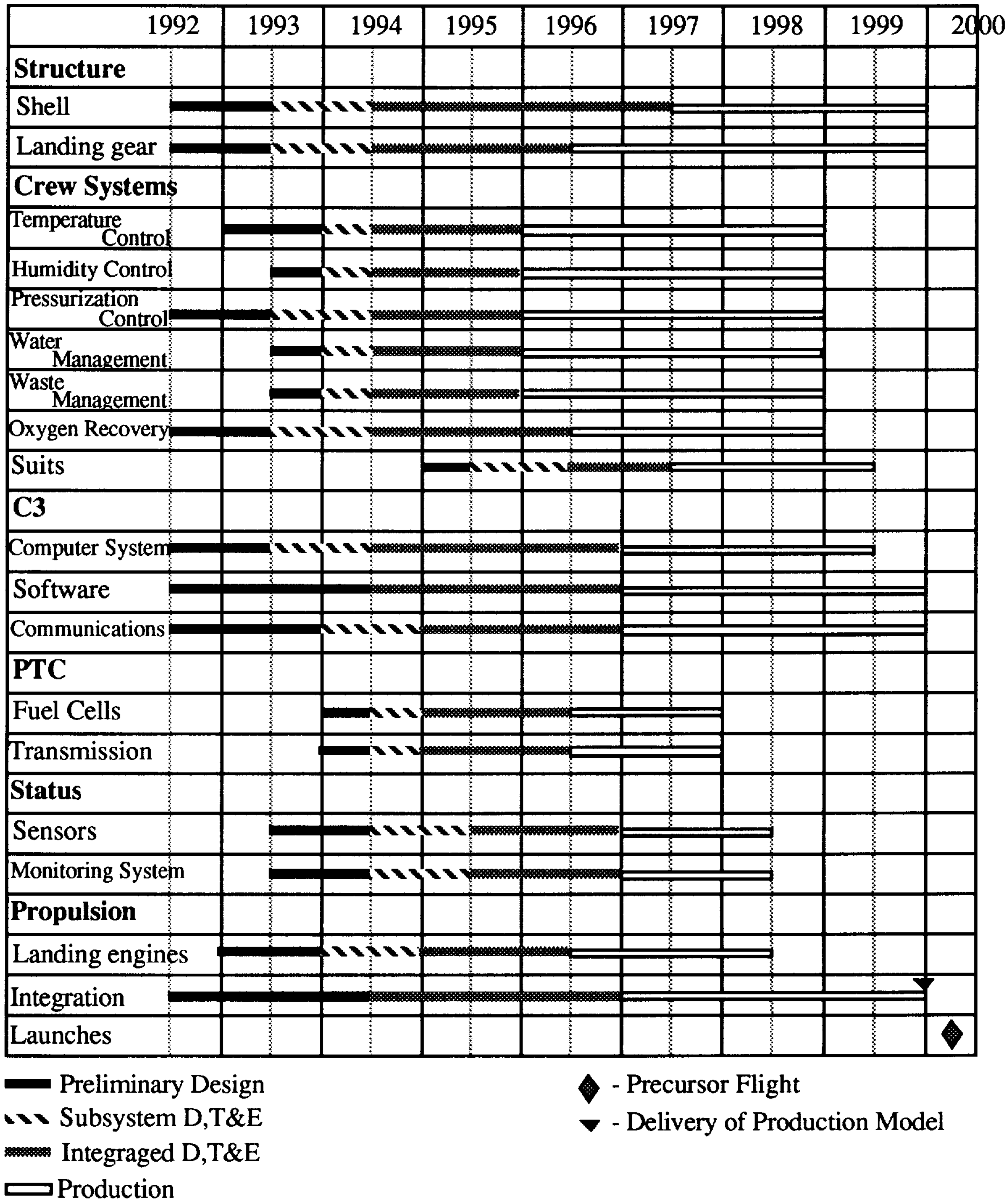

Figure 3-5

BioCan Development Schedule

3.2.3 PTLI Development Schedule

The development schedules of all the propulsive stages are very similar. They start in 1994 with a period of detailed designing. In 1996 the testing and evaluating process begins, and in 1998 the final production begins. The PTLI has a more developed GNC system than the other stages because of its solitary time in LEO. For the same reason, it is the only propulsive stage with its own power system. The PTLI will have two flight tests before qualifying as operational: the unpiloted test around the moon, and the first precursor mission. See Figure 3-6.

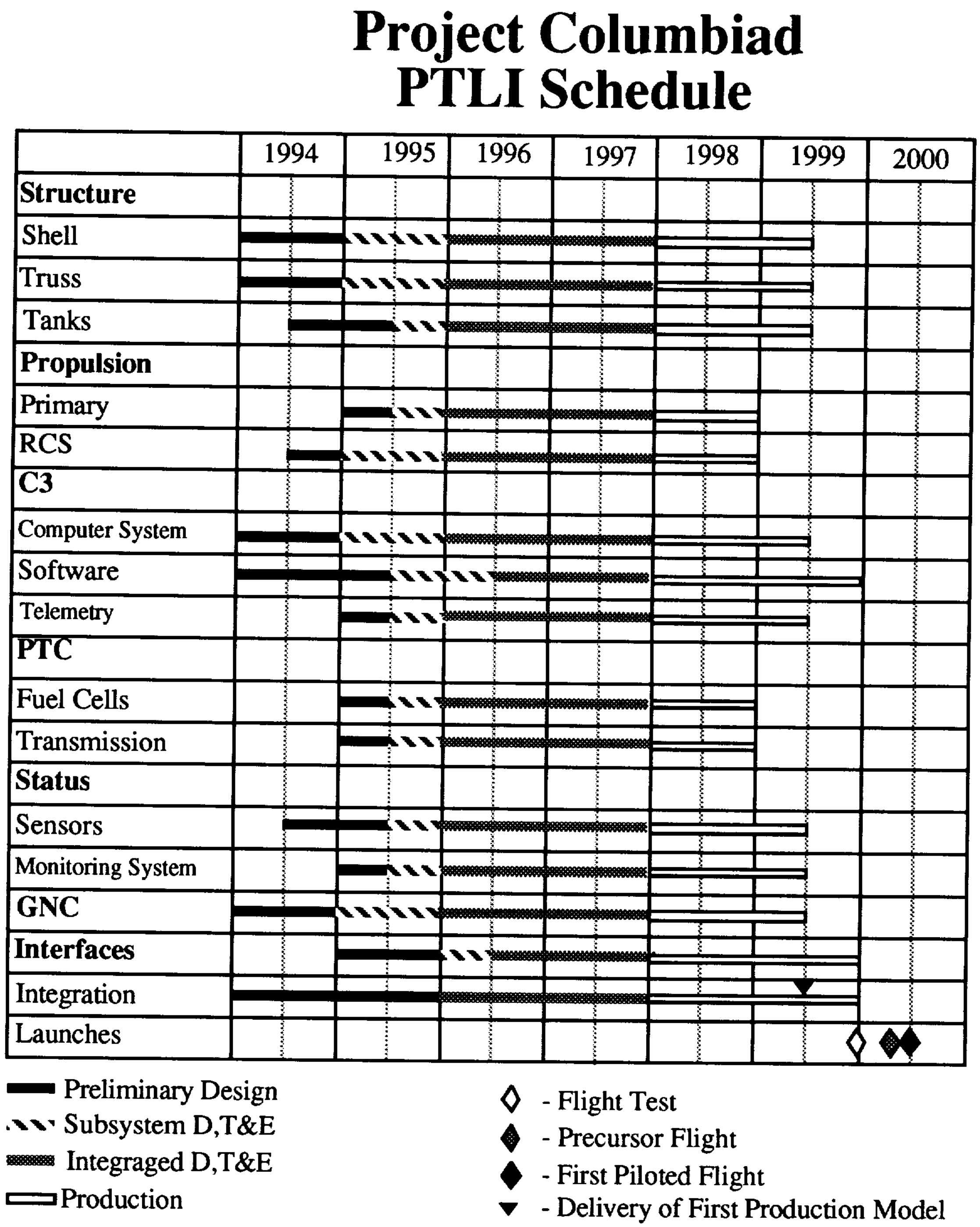

Figure 3-6
PTLI Development Schedule

3.2.4 LBM Development Schedule

The LBM development schedule is very similar to that of the PTLI. The largest and most important systems of the two, the structure and the propulsion systems, are nearly identical. In addition, the LBM has no computer or power systems. These functions are fulfilled by either the CM or the precursor mission payload. The detailed design process begins in 1994, the testing in 1996, and production in 1998. The LBM will also be tested twice before qualifying as operational. See Figure 3-7.

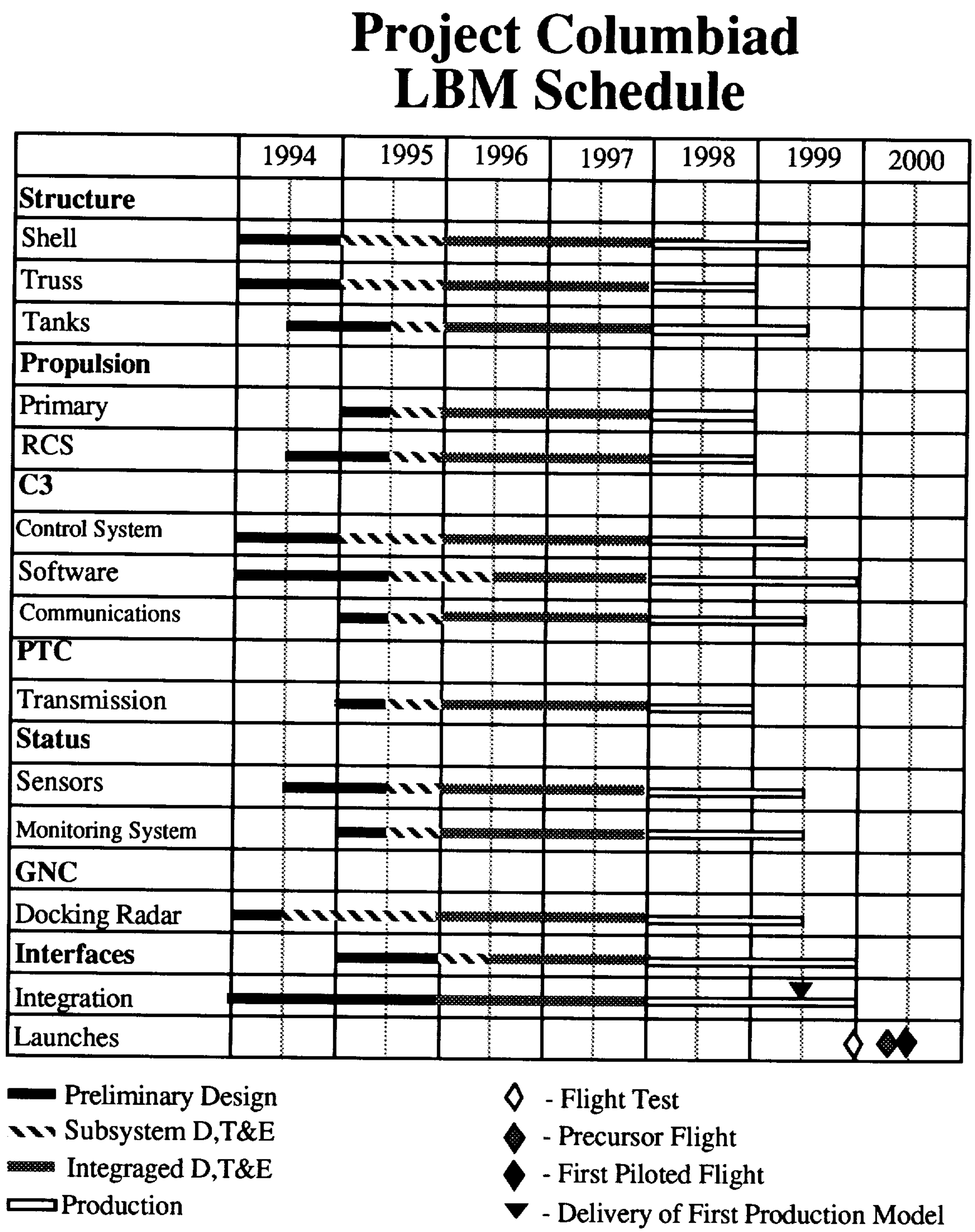

Figure 3-7

LBM Development Schedule

3.2.5 ERM Development Schedule

The ERM is similar to the LBM with the addition of landing gear, throttlable engines, fuel cells, and a payload storage capacity. The development schedule, therefore, follows closely that of the LBM. There will be only one test before it is declared operationally qualified. See Figure 3-8.

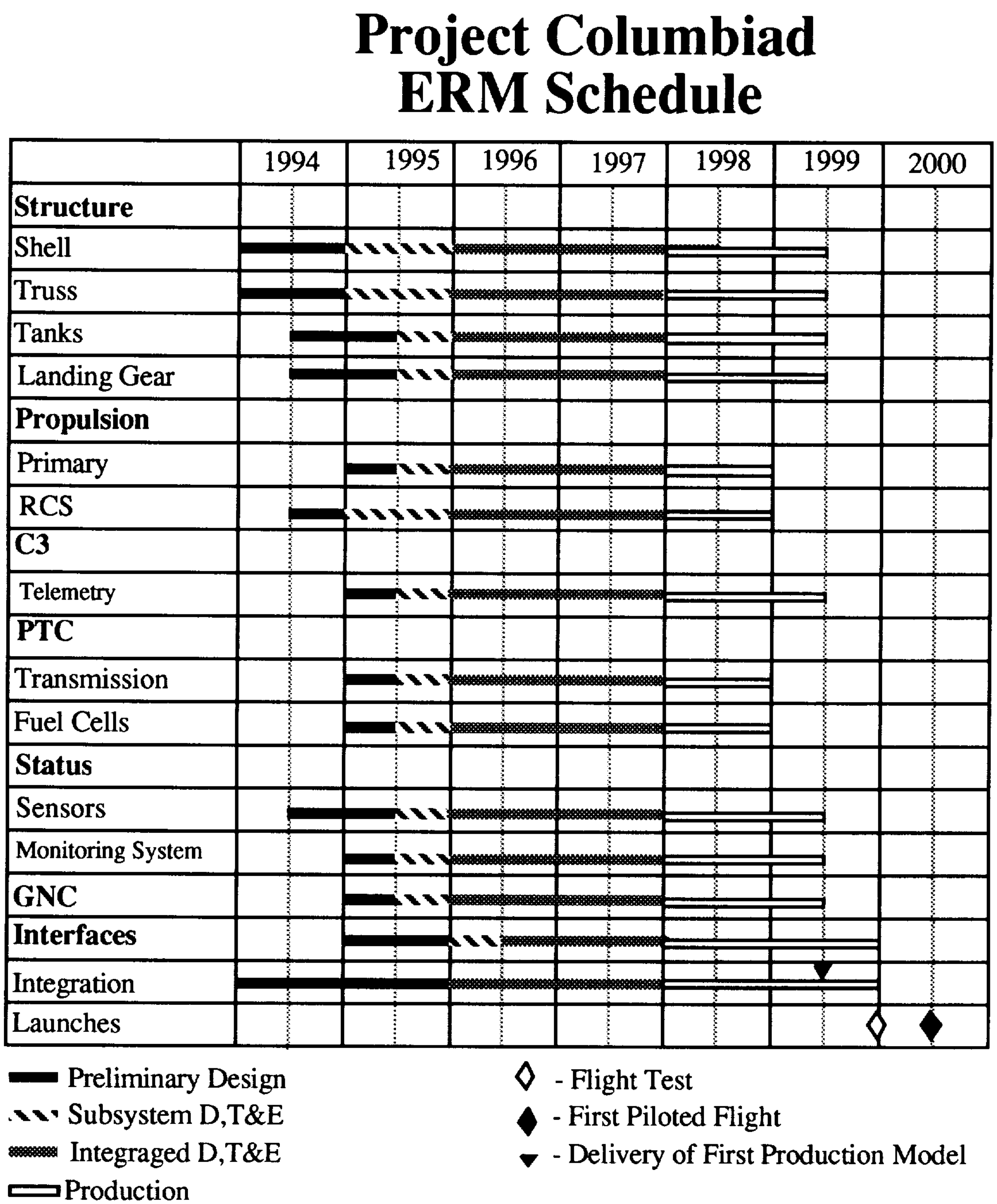

Figure 3-8
ERM Development Schedule

3.2.6 PLM Development Schedule

The PLM, meant to land on the moon and not lift off again, is the smallest of the propulsive stages. The propulsive system, however, is almost exactly that of the ERM. For this reason, and because the PLM will not be used for piloted missions, no test flights will occur before the first precursor mission in 2000. See Figure 3-9.

Project Columbiad
PLM Schedule

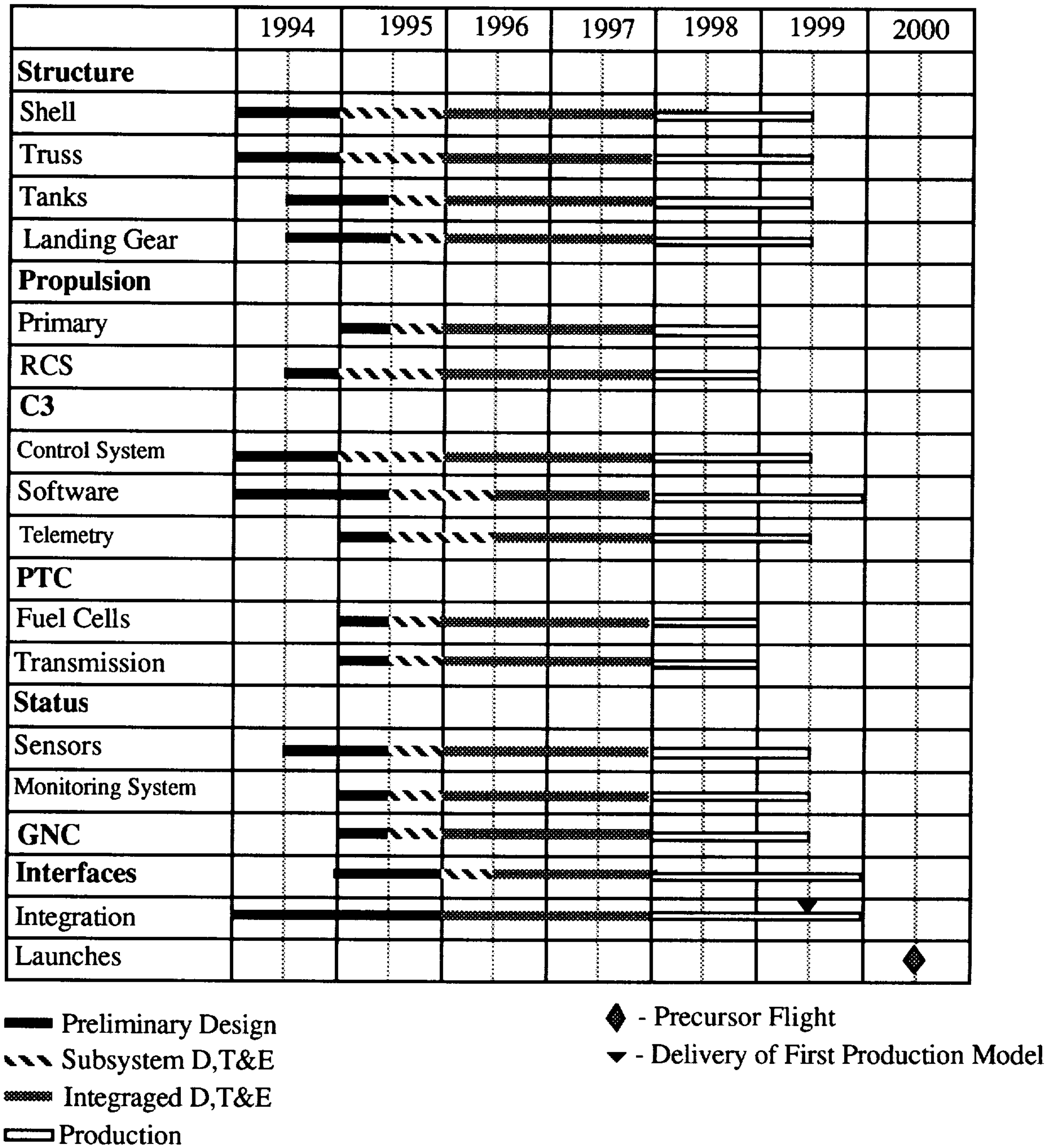

Figure 3-9

PLM Development Schedule

3.2.7 Surface Hardware Development Schedule

The surface hardware for Project Columbiad (excluding the BioCan) includes the rover, the bagger, and the Solar Lunar Power Plant. There development schedules are shown in Figure 3-10.

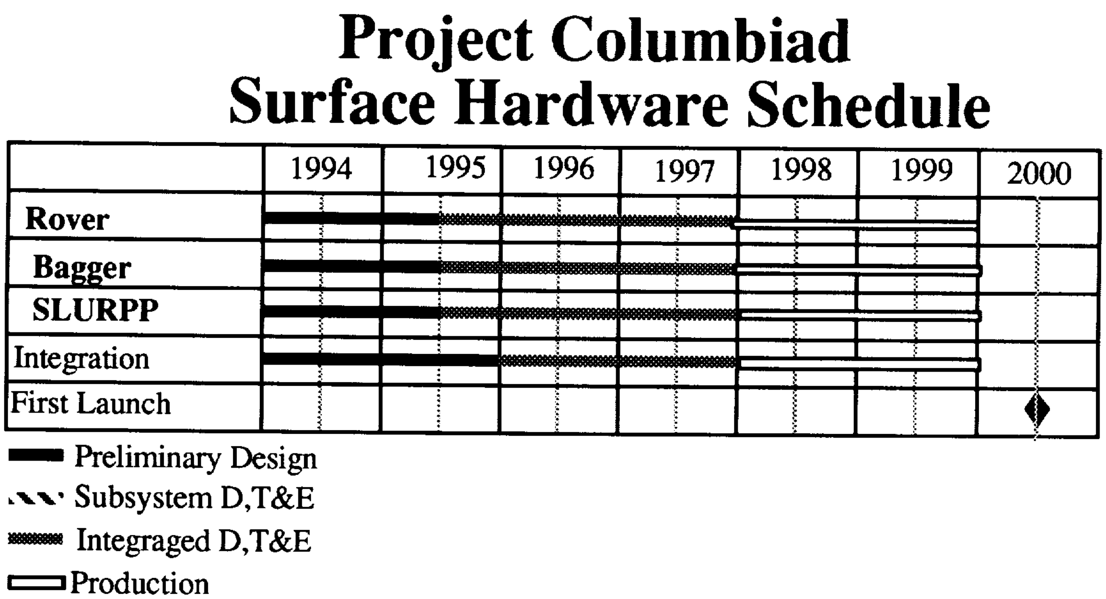

Figure 3-10
Surface Hardware Development Schedule

4 Campaign Strategy

4.1 Introduction

4.1.1 Why we need a clear strategy

Successful completion of Project Columbiad involves important engineering goals (demonstrating a renewed manned lunar capability, extended stay time, unrestricted landing zone, etc.); however, to ensure fiscal viability for the program, these goals must be balanced against additional scientific goals which will most certainly be raised by the scientific/academic community upon presentation of a $30-50 billion program to Congress. Project Columbiad's exploration/campaign philosophy is an attempt to balance these two (sometimes) competing forces.

Additionally, any program planning for Project Columbiad must be based on the assumption that no additional funding for hardware beyond that specified in the initial proposal will be readily forthcoming. As graphically demonstrated in various NASA programs (most notably, Apollo), expenditures for follow-on capabilities, even those acquired at a relatively modest cost, are rarely agreed to. Funding for programs which have already been approved or are in production or operation are more probable. Therefore, should Project Columbiad be funded it is imperative that all critical program goals be met using the initially allocated hardware.

4.1.2 Why go to the Moon

" WHY GO TO THE MOON? In one sense, this question is unanswerable, for the more astringent philosophers have found no completely convincing reason for any human activity, including breathing. But once it is admitted that life *is* worth living and that it was not all a dreadful mistake to have left those cozy Pleistocene caves, the question takes another form. We will obviously, because we are men, go to the moon as soon as it is technically and economically feasible." - Dr. Arthur C. Clarke, in Foreword of Neil Ruzic's "the case for going to the moon".

From an evolutionary point of view, the settlement and development of the Moon are inevitable in the process of human expansion. The process, anthropologists tell us, began somewhere in Africa less than a quarter billion years ago. Since the continental glaciers retreated about 12,000 years ago, human expansion over the Earth's surface has been well marked. The process continued through the discovery of the New Worlds, settlements in Australia and then in the first half of this century with the continuous occupation of

Antarctica. The Moon is the New World of our generation. Thirty years ago, when President Kennedy challenged the nation to land a man on the surface of the Moon by the end of that decade, and return him safely to Earth, the predominant motivation might have been a demonstration of political and technological supremacy. But today when we plan to go back to the Moon, in order to build a base for permanent presence on the Lunar surface, our reason is primarily intellectual. The Moon is the place for human beings to live and work in the 21st century.

Achieving higher levels of engineering prowess is of course still one of the main reasons for going to the Moon. Building superior spacecrafts, increasing the capabilities of the propulsion systems, exploring high-efficiency energy resources, developing new technologies for long-term human survival in hostile environment, advancing the state-of-art in space robotics for autonomous operations - all of the above leads to a multi-dimentional development of the state of technology on Earth. But besides that there are several scientific reasons to go to the Moon as well.

The Moon is a special museum where the early history of the Solar System is preserved. It is the key to understanding how all terrestrial planets were formed. the crust of the Moon has been differentiated by volcanism and the bombardment of meteorites and surface has preserved a record of all these activities since the first few million years of the Solar System history. The principal assessment of the nature and evolution of the Moon that exists today is mostly the product of four years of active Apollo exploration, supplemented by Soviet data. The analysis of the crater formation on the Moon led to the deduction of a massive bombardment of the inner Solar System. There are not enough data to tell how long this terminal cataclysm lasted. Future sample analysis may resolve this central question in understanding the formation of the planets.

Moreover, the surface of the Moon has remained so quiet that solar wind elements have accumulates undisturbed for eons, thus maintaining a record of the Sun's activity. Evidence of periodic variation has been found in the solar wind deposits on the lunar surface.Further investigation of the lunar rocks and dirt may lead to a reappraisal of our ideas about solar evolution.

The selenological and selenophysical explorations will also create an inventory of the lunar resources. Of particular importance are abundance of oxygen, metals, hydrogen (probably in the form of primordial ice or from solar wind deposits), and helium-3. This data will be very valuable not only for development of a self-sufficient Lunar base but for various benefits to Earth as well. Helium -3 is known as a potential source of clean energy when burned as fuel with deuterium in a thermonuclear fusion reaction. The isotope is rare on Earth (several hundred kilograms) whereas, the estimated mass of helium-3 accumulated on the lunar surface for billions of years as fallout from the solar wind is one million tons. As compared to fusion with tritium, the deuterium-helium-3 reaction is much cleaner and more efficient. It simplifies the safety-related design features of the fusion reactor. Extensive radioactive waste handling facilities are also eliminated. Using helium-3 as our primary source of energy for terrestrial needs is expected to be economical as well, the estimated yield from helium-3 fusion is 250 times more than the cost of energy to mine the isotope on the Moon, process it, and transport it to Earth. The same process of extracting helium-3 from regolith is likely to produce nitrogen, hydrogen and other elements. This hydrogen and oxygen, extracted from ilmenite and anorthite, in addition to lunar base uses, can also be used for fuelling future spacecrafts. In fact, helium-3 can also be used as astrofuel for nuclear powered spacecrafts.

The effectiveness of the Moon as a site for astronomical observations comes from various reasons, which include : the existence of a vacuum on the Moon along with a dark sky; the size and stability of the lunar surface for large baseline instrumentation; partial cosmic ray protection; a near-cryogenic temperature, particularly at the bottom of the craters in the lunar poles; slow rotation rate; and distance from Earth, and its electromagnetic environment. For these reasons, astronomers have dreamt for a long time, about lunar based astronomical observatories for long-term, multi-spectral coverage of space. Lunar poles and the far-side of the Moon present the best opportunities as a highly desirable location for placing major observatories in the future.

It is obvious that Project Columbiad's goals do not include achieving the above mentioned state of industrialization and development on the Moon. It is the job for a fully equipped and appropriately capable lunar base that is beyond the scope of this Project. However, it is necessary to keep this goals in the perspective, because they determine the nature of the Projects scientific experiments and exploration missions. This in turn will dictate the type of hardware we should be prepared to transport to the Moon in the future. Hopefully, with the positive results from the experiments conducted by Columbiad astronauts, we will be ready to begin industrialize the lunar base as our next step.

4.2 Landing site Selection

4.2.1 Selection Criterion

Project Columbiad's design conceivably allows for a choice of landing site at any longitude or latitude on the surface of the moon. Naturally, however, a significant portion of the lunar surface area must be excluded because of terrain features (craters, boulders, steep slopes, rilles, etc.) which prevent the safe landing of the spacecraft. Additionally, other operational constraints, including lighting, navigation, fuel consumption, etc., influence the desirability of certain landing sites over others. Finally, the scientific interests of the professional academic community (most importantly, the planetary science field) weigh in the decision over which landing site becomes the most desirable.

4.2.2 Apollo Example

Table 4-1 lists the landing sites which were recommended for exploration during the Apollo program (beginning with the *second* lunar landing, however, as the first lunar landing [Apollo 11] was less than two months away; that mission was planned to land in the safest and most boring location available, to fulfill the primary goal of Apollo as swiftly as possible). These sites were chosen to meet operational considerations (i.e., adequately smooth terrain for approach and landing, accessible to Apollo from a launch trajectory standpoint) and to satisfy certain scientfic criteria (mainly geological in nature). Only two of these sites were actually visited by Apollo astronauts; the remaining four Apollo landing sites were selected from alternate locations suggested later in 1969 and 1970. The Apollo 12 mission, for instance, was programmed to land in Oceanus Procellarum to rendezvous with the Surveyor III unmanned probe. The purpose of this selection was twofold: first, it allowed for the demonstration of a precision landing capability for the Lunar Module; second, the astronauts were able to bring back parts from the Surveyor for post-mission analysis (to determine what effects, if any, accompanied a 3-year exposure to the lunar environment). Later missions (Apollo 14-17) focused on visiting specific geologic formations of interest to scientists.

Table 4-1: Lunar Landing Sites Recommended for Consideration by the Apollo Site Selection board Meeting of June 3, 1969

Site	Latitude	Longitude
Censorinus	0°17' S	32°39' E
Rima Littrow	21°25' N	28°56' E
Abulfeda	14°50' S	14°00' E
Rima Hyginus	7°52' N	6°7' E
Tycho	41°8' S	11°35' W
Copernicus Peak	9°36' N	19°53' W
Copernicus Wall	10°22' N	19°59' W
Schröter's Valley	24°36' N	49°3' W
Marius F	15°10' N	56°31' W
Fra Mauro	3°45' S	17°36' W
Mösting C	1°55' S	8°3' W
Hipparchus	4°36' S	3°40' E
Prinz	25°57' N	43°40' W
Gassendi	17°50' S	40°20' W
Dionysius	2°31' N	17°49' E
Alexander	37°46' N	14°6' E
Alphonsus	13°35' S	4°11' W
Rima Bode II	12°47' N	3°49' W
Copernicus CD	6°32' N	14°58' W
Tobias Mayer P	13°18' N	31°11' W
Aristarchus	24°24' N	47°50' W

*Reproduced from NASA History Series SP-4214, p. 161.

Note: Fra Mauro was visited by the Apollo 14 astronauts in February 1971, and Littrow by Apollo 17 in December 1972.

4.2.3 Probable Sites

From these historical precedents, it is possible to suggest an exploration scheme for Project Columbiad. The choice of landing sites should reflect the following considerations:

- Early demonstration of precision landing capability for Project Columbiad hardware
- Demonstration of global landing capability

- Recovery of Apollo-era hardware for Earthside examination (30+ yr. lunar exposure time)
- Visitation of interesting geological sites as suggested by Apollo/Lunar Resources Orbiter data and scientific

Based on the above considerations, we have come up with a tentative list of sites for Project Columbiad Lunar Campaign. The details are discussed in the following section.

4.3 Proposed Campaign Plan

There are two basic approaches that can be taken towards lunar exploration under the Columbiad Lunar Campaign plan. One approach is to go to a different location in each of the five years. This means that at the end of the fifth year there will be five lunar outposts, almost identical in their capability. Each of these outposts will have been built up from the hardware of one precursor mission followed by three piloted missions. In order to maximize the exploration area, the outpost sites will be located around the lunar surface in varied selenological regions.

An alternative approach would be to dedicate the entire Columbiad campaign to initially set up one outpost in the most suitable site and then with each mission expand the capabilities of the outpost into a permanently occupied Lunar base. All the exploration will originate at this location and utilize improved second-generation surface vehicles. Remote sites can be visited during long excursions.

However, for Project Columbiad, a combined approach is more preferable. For the first three years, the first approach can be followed. This means three different sites will be visited each year and local features will be explored. At the end of the third year, one of the three sites will be chosen for expansion and revisited over the next two years. The fourth and fifth precursor will take different kind of hardware which will include a nuclear power plant, a pressurized heavy-duty surface vehicle and inflatable structures for habitat expansion. Thus experience gained over the first three years can be fed back into the campaign plan.

As a suggestion for the first site, we have selected the Lunar South Pole region. Detailed coordinates of the site, however, are unavailable at this time; the lunar polar regions have not yet been mapped in great enough detail. A Lunar Polar Orbiter Satellite is thus assumed to have been launched a few years prior to the beginning of Columbiad. The justification for going to the South Pole is that a site there can provide an unobstructed, continuous view of the center of our galaxy. In addition, all the reasons for going to the lunar poles mentioned in the previous section apply as well. One precursor followed by one piloted mission will set up the initial outpost and begin exploration of the area. The next piloted mission will continue with the exploration and begin other scientific missions. A significant portion of the second mission is likely to be devoted to astronomical experiments or setting up a small scale observatory.

The next precursor will be sent to the center of the lunar disc facing the earth. This outpost will revisit Apollo 15 or the Surveyor 6 site. One of the main tasks for the first piloted mission will be to analyze the effect of exposure from lunar environment on the hardware left behind thirty years ago. If the Hadley-Apennine (Apollo 15 site) is visited, one interesting aspect of the mission could be to revive the Lunar Roving Vehicle and perhaps even use it for additional transportation.

The third site could possibly be on the farside of the Moon. The feasibility of setting up an outpost on the farside depends on a Lunar Communication Satellite system, which possibly can be deployed, one satellite at a time, by all the previous missions. Crater Tsiolkovsky is a likely choice. Similar to the other two outposts, this site will also be visited by one precursor followed by two piloted missions.

In the fourth year, the second phase of the campaign will begin. A precursor mission carrying a heavy-duty vehicle will be sent. The vehicle (yet to be designed) should be able to combine the tasks of a bulldozer and a trailer-truck. It will have a pressurized habitat trailer equipped for a two-person crew for up to seven days. The expected range of such a vehicle is approximately 1500 kilometers. The piloted mission following the precursor will extensively use this vehicle for exploration away from the base. The next precursor will carry a nuclear reactor and the inflatable structures. The vehicle will assist in installing these equipments. The rest of the piloted missions will expand the base and their mission times are also likely to extend up to sixty days. The last two piloted missions would probably overlap their lunar stay to fully utilize the expanded facilities of the first Lunar base.

Table 4-2: Schedule for Five-year Campaign

YEAR	MISSION	SITE
2001	Precursor 1 Piloted 1 and 2	South Polar region
2002	Precursor 2 Piloted 3 and 4 Precursor 3	Sinus Medii 0.5° S, 1.4° W or Hadley-Apennine 26.1° N, 3.7° E Tsiolkovsky (?)
2003	Piloted 5 and 6 Precursor 4	Tsiolkovsky Selected site for Lunar Base
2004	Piloted 7 Precursor 5 Piloted 8 and 9	Lunar Base
2005	Piloted 10, 11, 12	Lunar Base

4.4 Beyond Columbiad

What lies beyond 2005? If Columbiad goes as per plan, then by that year humankind should have their first home beyond their home planet . Soon after, Helium-3 production and transportation to Earth will probably start paying off the investment of building the base. The experience of building a permanently occupied, self-supporting Lunar base would be very valuable in planning a Mars expedition in the next ten years. Between 2005 and 2019, the Lunar base will probably be sufficiently industrialized to be a major resource for fuelling Mars-bound spacecraft. Moreover, the Lunar terrain would probably be used for field testing all the hardware designed for the Mars campaign. The human imagination will be filled with the anticipation of visiting new regions of space where no one has gone before. We surely believe that is where the future of humankind is leading towards. Ad Astra.

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APPENDIX I - Subsystem Cost Estimations

All costs in $Millions

Crew Module

Piloted (CM)					Precursor				
name	**cost**	**no.**	**tot cost**	**TU**	**name**	**cost**	**no.**	**tot cost**	**TU**
IVA overboots	0.05	4	0.2	2.5	vacuum	0.01	1	0.01	1
IVA ovgarmen	5	4	20	4	EVA test stat	10	1	10	1
IVA suit	1	4	4	2o4	EVA hardsuit	8	5	40	3
shoes	0	4	0	1	shoes	0	4	0	1
uniforms	0.0015	1	0.0015	1	uniforms	0.0045	1	0.0045	1
sleepers	0.0005	4	0.002	1	sleepers	0.0005	4	0.002	1
food	0.0005	1	0.0005	1	food	0.00204	1	0.00204	1
med kit	0.00008	1	0.00008	1	med kit	0.00015	1	0.00015	1
LOX tank	0.01	3	0.03	2	LOX tk	0.01	3	0.03	2
N2 tank	0.01	3	0.03	2	LN2	0.01	3	0.03	2
drink H2O tk	0.01	2	0.02	1	drink H2O tk	0.01	6	0.06	1
commode	0.2	1	0.2	1	commode	0.2	1	0.2	1o2
Hum Control	1	1	1	1o2	Hum Control	1	1	1	1o2
piping	0.1	1	0.1	2	H2O recovery	1.5	3	4.5	1o2
LiOH system	2	1	2	1	recovery tk	0.01	1	0.01	
Heat Control	2	1	2	1	piping	0.6	1	0.6	2
Mass Spectro	0.5	1	0.5	1	LiOH sys	2	1	2	1
Gas Control	included	1	0	1	molec sieve	1.5	2	3	1
tubing	0.1	1	0.1	1	heat contol	2	1	2	1
smoke detect(	0.0012	3	0.0036	1	mass spectro	3	1	3	1
area sm detec	0.0035	3	0.0105	1	gas control	included	1	0	1
flame detec	0.00875	3	0.02625	1	electrolysis	included	1	0	1o2
lighting	0.005	1	0.005	1	sabatier burn	included	1	0	1o2
tool & clean	0.003	1	0.003	1	tubing	0.6	1	0.6	1
multichan ECG	0.005	1	0.005	1	smoke detecto	0.0081	20	0.162	1
Exerciser	0.00015	1	0.00015	1	area sm detec	0.0094	8	0.0752	1
					flame detecto	0.0146	5	0.073	1
TOT COST			**30.24**		lighting	0.011	1	0.011	1
					tools & clean	0.01	1	0.01	1
Set Costs			5.98		Treadmill&bik	0.0236	1	0.0236	1
Prod Costs			24.26		Metabolic anal	0.015	1	0.015	1
					TOT COST			**67.42**	
					Set Costs			16.81	
					Prod Costs			50.61	

Communications

SET MANUFACTURER'S PRICE					
Component	**Cost**	**Number**	**Total Cost**	**TU**	
RH32 Data Processor	0.01	7	0.07	1	
Fairchild Solid State	0.001	5	0.005	1	
Universal Demodulator	0.001	14	0.014	1	
High Data Rate Modem	0.001	8	0.008	1	
Antenna Pointing Syste	0.01	3	0.03	1	
High Gain Antenna	0.002	6	0.012	1	
Low Gain Antenna	0.002	16	0.032	1	
Receiver	0.005	30	0.15	1	
Transmitter	0.004	14	0.056	1	
Power Supply HP	0.015	3	0.045	1	
Inter-Comm System	0.004	1	0.004	1	
Telephone System	0.004	1	0.004	1	
Video Camera	0.001	2	0.002	1	
Microphone	0.0002	2	0.0004	1	
2-Way Multiplexer	0.00025	15	0.00375	1	
Amplifier	0.002	14	0.028	1	
Cable/Fiber Optics Line	0.0005	22	0.011	1	
Stub Tuner	0.00025	22	0.0055	1	
Switch/MUX	0.0001	7	0.0007	1	
Data Bus (Software)	2	1	2	1	
DSN Use	0			1	
TOTAL SET COST			**2.48135**		
PRODUCTION COST					
Component	**TFU Cost**	**Number**	**RDTE Cost**	**TU**	**Total Prod. Cost**
Computer	0.9	6	2.7	2.5	5.4
MDM-16 MUX/DEMUX	0.204	2	0.612	2.5	0.408
Odetics Tape OHSR	0.101	2	0.303	2	0.202
TOTAL PROD COST			**3.615**		**6.01**
MODULE BREAKDOWN					
Crew Module	**2.852**				
Earth Return Module	**0.032**				
Habitat	**2.862**				
PTLI Stage	**0.286**				
Rover	**0.069**				

Guidance and Navigation

SET PRICES					
Name	**Cost**	**Number**	**Total Cost**	**TU**	
Sun Sensor	0.1	3	0.3	1	
Earth Sensor	0.1	2	0.2	1	
Star Tracker	0.5	6	3	1	
GPS Receiver	0.004	6	0.024	1	
Liq Cryst Disp	0.004	3	0.012	1	
Joysticks	0.002	2	0.004	1	
Radar Alt.	0.05	6	0.3	1	
Rubid. Clock	0.125	1	0.125	1	
Ant. Beacons	0.015	2	0.03	1	
TOTAL SET COST			**3.995**		
PRODUCTION COSTS					
Component	**TFU**	**Number**	**Total Cost**	**TU**	**RDTE**
INS	0.01	6	0.06	2.5	0.03
Dock Vid Cam	0.01	1	0.01	2	0.03
Las Dock Rada	0.1	1	0.1	3.5	0.4
TOTALS			**0.17**		**0.46**
MODULE BREAKDOWN					
PTLI	**0.333**				
LBM	**0.11**				
ERM	**1.78**				
CM	**0.054**				
Precursor	**1.888**				

Power

Production Costs (millions)				
w/ 6 prototypes				
Component	**cost**	**number**	**Total Cost**	**TU**
Fuel Cells	13.4	1	13.4	
Electrolysis Cells	18.4	1	18.4	
Radiators	15.9	1	15.9	
Solar Array	11.6	1	11.6	
Oxygen Tanks	19.9	1	19.9	
Oxygen Tank Lining	0.4	1	0.4	
Hydrogen Tanks	30.9	1	30.9	
Hydrogen Tank Lining	0.4	1	0.4	
Water Tank	3.6	1	3.6	
Integration	3.6	1	3.6	
Li battery pack	0.1	1	0.1	
Cryogenics&coating	6	1	6	
MODULE BREAKDOWN				
Surface Payloads	118.1			
Crew Module	0.1			
ERM	18.86			
PLM	12			
LBM	6			
PTLI	18.86			